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USAAVLABS TECHNICAL REPORT 69-90

INVESTIGATION OF CH-54A ELECTROSTATIC CHARGING AND OF ACTIVE ELECTROSTATIC DISCHARGER CAPABILITIES

By

Michael C. Becher

January 1970

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DAAJ02-69-C-0102
DYNASCIENCES CORPORATION
SCIENTIFIC SYSTEMS DIVISION
BLUE BELL, PENNSYLVANIA

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This report was prepared by the Dynasciences Corporation, Scientific Systems Division, under Contract DAAJ02-69-C-0102. The effort was undertaken at the request of the CH-54A Project Manager's Office. The basic objectives of the effort were to determine the maximum charging current levels that might be experienced by a CH-54A and to determine the capability of the Dynasciences D-04 and D-15 active electrostatic dischargers to compensate for them.

The results of the test program show that the CH-54 operating in a sand and dust environment will generate charging currents greatly in excess of the discharging capability of the tested systems. It can be seen that a substantial amount of research and development effort must yet be accomplished to solve the overall problem. For the short-term solution to the CH-54A problem, it appears that the use of dual high-voltage units of a slightly refined design will suffice. Such an installation would be capable of discharging approximately 200 microamperes, which will satisfactorily discharge the aircraft in a large percentage of its operating conditions.

The conclusions and recommendations contained in this report are concurred in by this command.

Project 1X163203D332
Contract DAAJ02-69-C-0102
USAAVLABS Technical Report 69-90
January 1970

INVESTIGATION OF CH-54A ELECTROSTATIC CHARGING AND OF ACTIVE ELECTROSTATIC DISCHARGER CAPABILITIES

Final Report

By

Michael C. Becher

Prepared by

Dynasciences Corporation
Scientific Systems Division
Blue Bell, Pennsylvania

for

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

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SUMMARY

A test program was conducted at Yuma Proving Ground, Arizona, to obtain in-flight measurements of electrostatic charging rates experienced by a CH-54A helicopter operating in a dusty environment and to evaluate active electrostatic discharger systems as solution techniques. This report presents data obtained during this program and reviews these data along with those from previous work on the same aircraft type. This report concludes that an extremely high and possibly lethal charge level is present on the CH-54A in the operational hookup situation, that an active discharger system capable of discharging 200 microamperes is required to dissipate this charge, and that, while presently available equipment does not meet this requirement, repackaging of present hardware offers a probable solution.

FOREWORD

The flight test program reported herein was conducted by the Scientific Systems Division of Dynasciences Corporation under the sponsorship of the United States Army Aviation Materiel Laboratories, Contract DAAJ02-69-C-0102, Project 1X163203D332. Mr. S. Blair Poteate, Jr., was the Army Project Engineer; Mr. Michael C. Becher was the Scientific Systems Division Project Engineer, assisted by Mr. Warren M. Exmore.

The author gratefully acknowledges the active participation of Mr. Poteate in the planning, installation, and flight phases of the program.

Mr. Ronald Spicer was present at the test site in the capacity of observer for an interested agency, the United States Army Electronics Command. The author is indebted to Mr. Spicer for his considerable contributions to the engineering and fabrication work on the exciter/multiplier fixture and for his photographic record of the aircraft hovering in a dust cloud.

The assistance of Mr. Joseph Dupnik, observer for Sikorsky Aircraft, in the installation phase of the program is acknowledged and appreciated.

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INTRODUCTION

Reports from operations in Southeast Asia indicate that a serious problem exists relative to electrostatic charge accumulation on heavy-lift helicopters and resultant personnel shocks.

CH-54 helicopters are experiencing a high incidence of electrostatic charging, most frequently when the aircraft is hovering in dusty areas while sling loads are being attached. The electrical shock received by hookup men is sufficient to knock them off their feet when they touch the helicopter's cargo hook.[1] Another frequently reported complaint is that the hookup personnel receive severe hand injuries as a result of electrostatic discharge: the edges and tips of their fingers and nails are split.

In January 1969, USAMC responded to an ENSURE requirement for a static discharge system to be used on the CH-54A helicopter, AMC directed that a test program be conducted to define the capabilities of the specified device, a Granger Associates passive discharger (P-STAT) system. In addition, AMC wanted to evaluate the feasibility of the Dynasciences Corporation active electrostatic discharger (ESD) system as a solution to electric charging of the CH-54A aircraft.

AMC required the following information:

1. A definition of the problem, in terms of potential shock hazards from electrically charged CH-54A aircraft.

2. An evaluation of the performance of each proposed solution relative to an unprotected aircraft.

3. An evaluation of the performance of the two proposed solutions relative to each other; that is, the passive discharger versus the active discharger.

To satisfy these requirements, a test program was carried out at Lakehurst Naval Air Station, New Jersey, during February and March 1969.

That test program showed that the CH-54A helicopter would develop what has been defined as a lethal electrostatic charge under conditions wherein the natural charging rate exceeded 6 microamperes. It also demonstrated clearly and conclusively the feasibility of the active ESD system for use in discharging the CH-54A helicopter, when charging currents experienced are within the operating range of the system. In addition, the test results showed that the passive discharger can offer no benefit under those conditions of most serious concern; namely, charging environments in which the rate exceeds 10 microamperes.[2]

While the Lakehurst tests provided a first step in studying and solving the problem of electrostatic charging of the CH-54A, they by no means constituted a complete program, for the following reasons:

1. The great bulk of the testing was carried out using artificial charge inducement. Whereas the test results clearly indicated the relation of shock hazard to the level of charging current, they did not indicate what levels of charging current are likely to be encountered during typical operations under natural charging conditions. The natural charging current at Lakehurst was minimal: about 1 microampere throughout the entire 6-week program.

2. The technique of artificial charging employed at Lakehurst was not a valid simulation of the natural charging environment for testing an active ESD system.

3. The Lakehurst test program did not define the operating limits of the ESD system.

The ESD maintained the aircraft's stored energy far below the hazardous level and the level at which the P-STATS operated, while it experienced charging at rates within the operating range of the particular system used. Two different ESD models were tested: the D-04, rated at 50 microamperes maximum discharge, and the D-15, rated at 150 microamperes maximum discharge. But the purpose of the evaluation of the ESD system at Lakehurst was to determine its feasibility as a solution to the problem of static charge accumulation on the CH-54A; it was not intended that this program define the operating limits of either the D-04 or the D-15 system, nor did it provide for optimization of the ESD component locations on the aircraft.

An optimized multiplier location is of great importance, but such a location was not established during the first test program. However, the data from that test series indicated that the D-04 system could be completely optimized on the CH-54A helicopter; that is, it could be made to provide a net discharge of nearly 50 microamperes. It appeared that, through proper location of the multiplier units, the D-15 could be set up to provide a net discharging capability of nearly 100 microamperes on this aircraft without a need to resort to more sophisticated and experimental optimization techniques.

A second program was instituted by USAAVLABS, in response to a request from the CH-54 Project Manager, to take up where the first tests ended:

1. Test the helicopter in high-natural-charging-rate environment to determine what charging currents it can experience in an operational situation.

2. Define the best aircraft location for the ESD multiplier and determine the discharging capability of each system on the CH-54A.

This program was conducted at Yuma Proving Ground, Arizona, during the period June 12 through June 21, 1969. The following is a detailed report on that program.

DISCUSSION

TEST PROGRAM

Problem Definition Tests

Two types of tests were performed to define the electrostatic charge problem on the CH-54A aircraft. Some information regarding the nature of this problem is presented in Appendix III.

The first tests were performed to measure the charging current levels experienced by the CH-54 performing normal cargo hookup operations (approach and hover) in a dusty environment. Basically, these tests involved a measurement of the natural charging rate while the helicopter was hovering. It had been intended to measure the voltage levels and compute the energy levels accumulated as well. However, the loss of some of the test instrumentation while enroute to the test site prevented the accumulation of aircraft potential data needed for calculation of energy levels.

The aircraft-to-ground capacitance was also measured during this test program. While this information had been obtained during the earlier test series at Lakehurst,[2] the accumulation of this data did not involve much in the way of time or effort and was repeated at Yuma for correlation with the tests at Lakehurst.

The instrumentation and procedures employed for the problem definition tests are described in Appendix I.

ESD Evaluation

This phase of the program had two goals:

1. To determine where to locate the ESD exciters and multipliers on the CH-54A helicopter in order to effect maximum discharging efficiency in a standing hover.

2. To determine the maximum net discharging capability of each of the ESD systems tested.

In general, the multiplier units should be situated in the area of maximum rotor downwash in such a way that there are no parts of the helicopter directly downwind of the discharge probes.

The optimization test series consisted of measurements of ESD system discharging capability conducted with different exciter/multiplier component locations. The purpose of the discharging capability tests is to obtain data regarding the efficiency of the high-voltage multipliers as dischargers. The overall efficiency of the system is dependent upon optimization of the multiplier location for maximum discharge with minimum recirculation.

The instrumentation and test procedures for this phase of the program are described in Appendix I.

TEST RESULTS

Problem Definition Tests

It was the intent of these tests to determine the most severe charging current conditions which might be encountered by the CH-54A in a somewhat typical adverse environment, more specifically to measure the charging current generated with the aircraft operating in the dense sand and dust cloud generated by an approach and hover over sandy terrain.

Inspection of the test site, designated Phillips Drop Zone, prior to the initiation of the tests revealed that a surface crust had been formed by recent rainfall. A jeep was driven over the area in a crisscross pattern, in order to break up the crust and provide a surface which was generally loose and sandy.

On the first flight into the test site, grounding of the aircraft and instrumentation was accomplished by means of a drop line, of approximately 100 feet in length, with a metal weight secured to the end which touched the ground. As the aircraft descended to an altitude of approximately 100 feet, the drop line made contact with earth ground and maintained ground contact as the aircraft flared and hovered.

The instrumentation was set up such that the drop line current could be both visually observed on microammeters and permanently recorded by means of a strip chart recorder. However, upon entry into the test conditions, it was observed that the recorder was inoperative and no alternative means was immediately available to tabulate the charging currents experienced. During the course of this test, which lasted approximately 2 to 3 minutes and involved a flare and hover at about 20 feet, very high charging currents were experienced. The magnitude of the charging currents was observed by Army and contractor personnel in the aircraft to vary between extremes of approximately 100 and 325 microamperes. The charging current appeared to have a center value on the order of 200 microamperes.

For subsequent flights into the dusty area, the test procedure was modified in an attempt to achieve a better electrical connection between the aircraft and earth ground.

The metal weight at the end of the drop line was eliminated. Before any measurements could be made, the aircraft was landed at the test site. A stainless steel rod was driven into the ground, and a bucket of water was dumped in the immediate area (within a radius of about 1 foot of the rod) to wet the ground adjacent to the rod. The drop line was attached to the rod. The helicopter was then raised to a 15- to 40-foot hover while the drop line current was recorded.

During these tests, the recorder was intermittently operative, but data was manually tabulated as a backup. Results of natural charging current measurements made during these test series are presented in Figure 1 and in Tables I and II.

Figure 1 is a chart recording of the natural charging current, showing its magnitude and variation with time. On this record, a high reading in excess of 150 microamperes is followed about five seconds later by a low reading of approximately 25. While the magnitude of the charging current in Figure 1 is considerably lower than that experienced during the first flight test over the dusty terrain, the variation with time is typical of experience throughout the test program.

Table I contains a series of natural charging current observations made while the test aircraft was hovering with no load over dust (same test run which yielded the chart recording in Figure 1). The charging current readings in Table II were made while the test aircraft was carrying a 6000-pound load and hovering over dust.

The high and average charging current readings obtained during succeeding flights over the dusty test site were consistently lower than during earlier ones. The data in Tables I and II illustrate this point. There appear to be two reasonable explanations for this reduction in charging current.

Experience in similar flight test programs has indicated that, in a given environment, the charging current is greater for a heavier aircraft.[3] However, the Yuma test runs yielding the data in Tables I and II contradicted earlier experience.

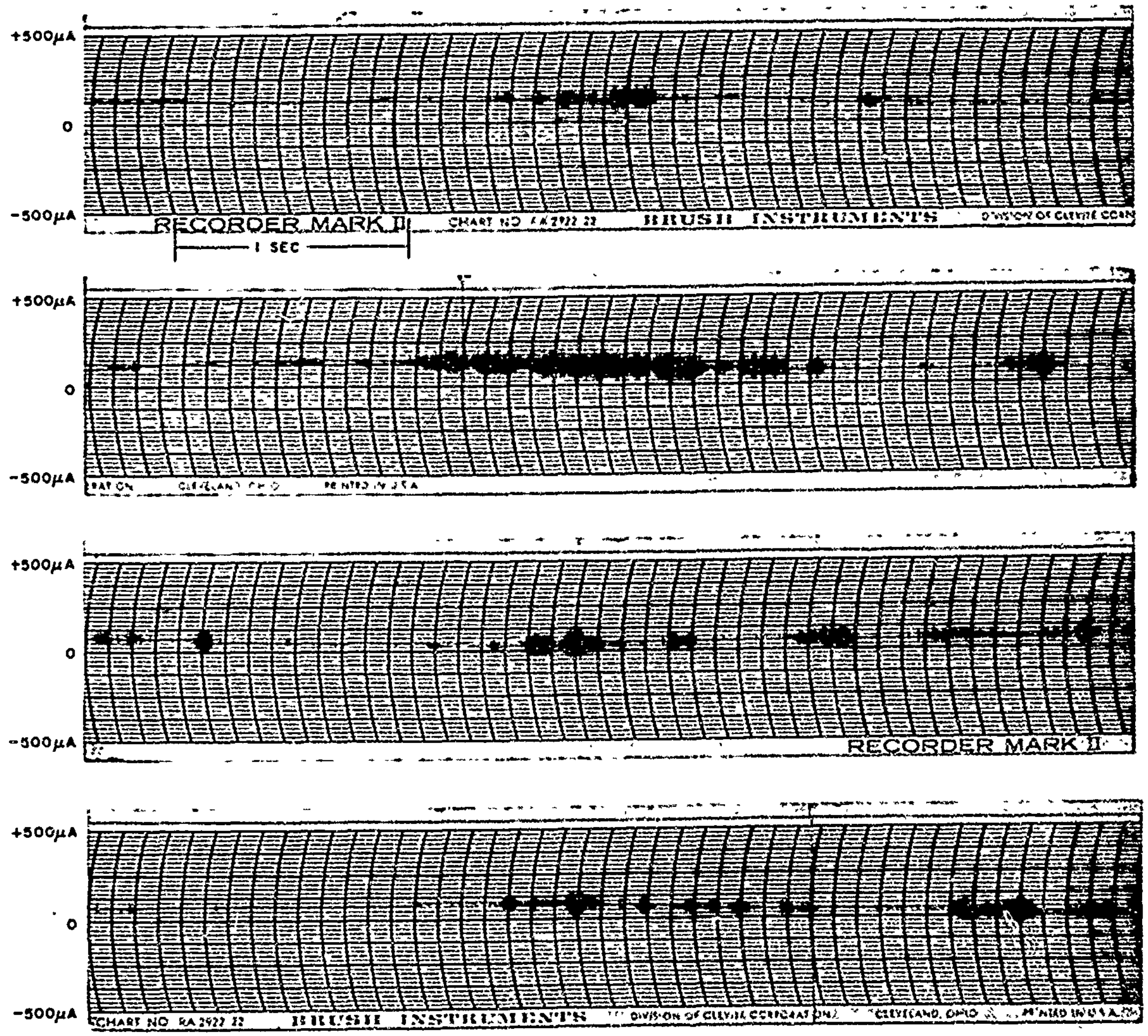

Figure 1. Chart Recording of Natural Charging Current Experienced by CH-54A Hovering Over Dusty Terrain (Aircraft Gross Weight Approximately 29,000 Pounds).

TABLE I. OBSERVATIONS OF NATURAL CHARGING CURRENT

	I_N (μA)		I_N (μA)		I_N (μA)
1.	190	12.	35	23.	125
2.	220	13.	34	24.	150
3.	120	14.	38	25.	125
4.	110	15.	40	26.	75
5.	100	16.	48	27.	70
6.	75	17.	55	28.	75
7.	60	18.	60	29.	85
8.	70	19.	80	30.	80
9.	25	20.	90	31.	100
10.	35	21.	100	32.	97
11.	25	22.	100		

Remarks: Readings were recorded continuously in the sequence shown, at approximately 4- to 5-second intervals.

The test helicopter was hovering with no load (gross weight approximately 29,000 pounds) over dust (Phillips Drop Zone).

TABLE II. OBSERVATIONS OF NATURAL CHARGING CURRENT

	I_N (μA)		I_N (μA)		I_N (μA)
1.	30	16.	70	31.	40
2.	40	17.	100	32.	35
3.	50	18.	125	33.	50
4.	60	19.	150	34.	60
5.	80	20.	100	35.	70
6.	100	21.	75	36.	80
7.	75	22.	100	37.	20
8.	50	23.	100	38.	12
9.	20	24.	125	39.	8
10.	25	25.	100	40.	15
11.	40	26.	75	41.	60
12.	60	27.	70	42.	80
13.	60	28.	25	43.	70
14.	80	29.	30	44.	60
15.	70	30.	35	45.	50

Remarks: Readings were recorded continuously in the sequence shown, at approximately 4- to 5-second intervals.

The test helicopter was hovering with load (gross weight approximately 35,000 pounds) over dust (Phillips Drop Zone).

leading to the conclusion that the environment was not the same for both runs; that is, the repeated flying and hovering over the same area tended to dissipate the dust to some extent.

This conclusion appears sound in view of the considerable dust cloud development that was experienced (see Figures 2 through 5). The loosely packed sand and dust was disturbed by the rotor downwash, picked up by the air recirculating through the rotor system, and formed into a dense dust cloud. The cloud was so dense, in fact, that the aircraft was barely visible from any point outside the dust cloud. The photograph sequence in Figures 2 through 5 shows the helicopter almost indiscernible as it hovered over the dusty area (Figure 2), slightly more visible as it began to emerge (Figure 3), and then standing out quite clearly against the background of the dust cloud it had developed (Figures 4 and 5). The dust cloud extended about 100 feet into the air, and it continued to rise above the test site after the helicopter had flown back to Laguna Army Airfield and landed (some 3 or 4 minutes after leaving the area).

A second explanation relates to the test procedures used. It is possible that merely lifting the helicopter to a given hover altitude over loose sand and dust produces a lower charging current than does an approach, flare and hover over that same terrain. On the basis of prior tests conducted by AVLABS, it appears likely that the higher downwash velocities experienced during the flare and transition to hover produce a higher-density dust cloud which can subsequently be maintained by the normal hover downwash velocities.

Since the purpose of measuring charging current was to determine what charging currents the CH-54 will experience in an operational situation, the data contained in Tables I and II were subjected to statistical analysis to more clearly define the problem and the requirements for a solution. Using the methods outlined in Appendix IV, these data were analyzed to arrive at a decision regarding the probability of experiencing a charging current of any given magnitude.

The data points are assumed to be random samples from an infinite population. It is easy to realize that the population is infinite. It can also be asserted with some confidence that the observations constitute random samples, in that no bias (other than the selection of a dusty environment rather than a random environment) was deliberately introduced to the data-taking. For instance, no considerations based upon the values of the test data entered into decisions as to when to begin or end data-taking during a particular run.

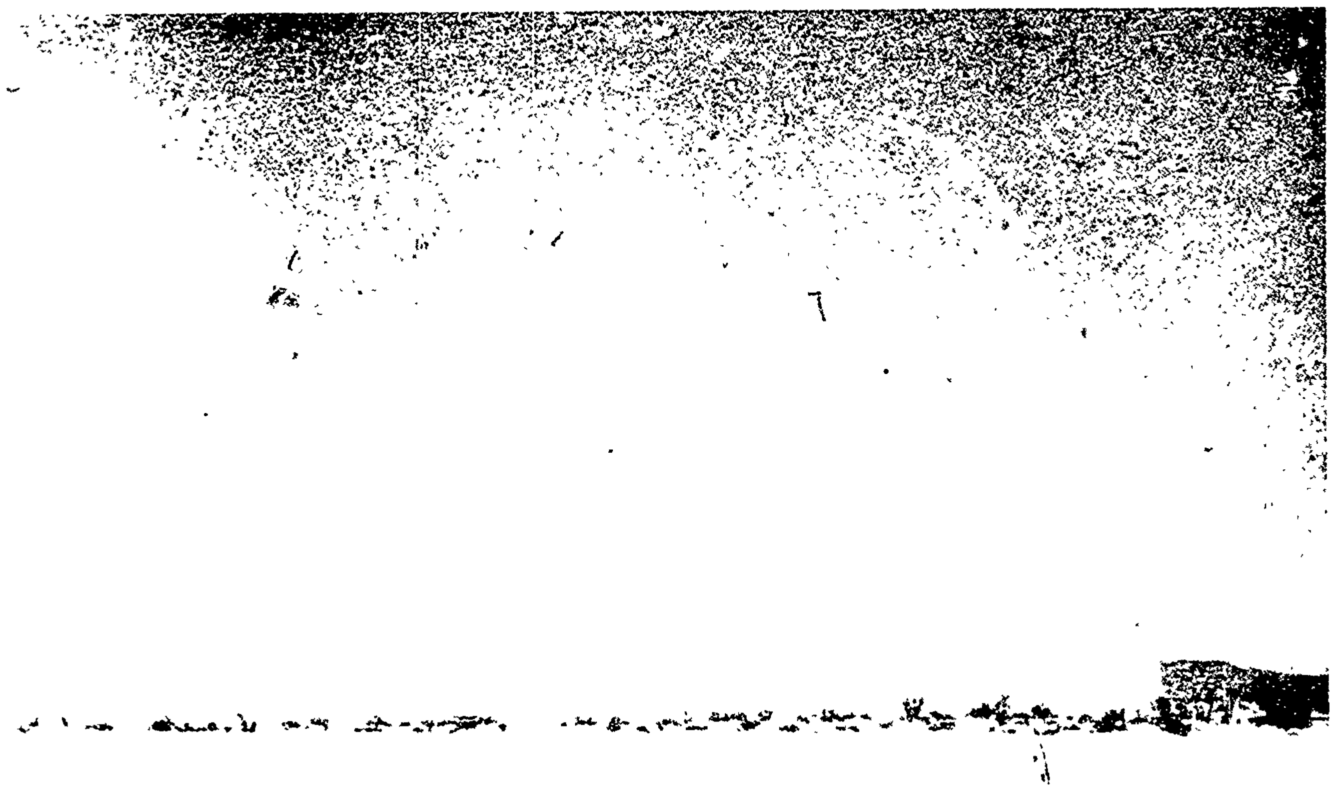

Figure 2. CH-54A Test Aircraft Hovering in Dust Cloud (Phillips Drop Zone, Yuma Proving Ground, Arizona).

Figure 3. CH-54A Test Aircraft Emerging From Dust Cloud (Phillips Drop Zone, Yuma Proving Ground, Arizona).

Figure 4. CH-54A Aircraft and Dust Cloud (Phillips Drop Zone, Yuma Proving Ground, Arizona).

Figure 5. CH-54A Test Aircraft and Dust Cloud (Phillips Drop Zone, Yuma Proving Ground, Arizona).

The data set in Table I will be considered first. The sample mean $\bar{x}$ is determined as follows (the units--microamperes--are omitted from the calculations of the statistical parameters):

$$\bar{x} = \frac{190 + 220 + \ldots + 97}{32} \tag{1}$$

$$\bar{x} = 84.12 \tag{2}$$

The sample variance s^2 is given by

$$s^2 = \frac{(190-84.12)^2 + (220-84.12)^2 + \ldots + (97-84.12)^2}{31} \tag{3}$$

$$s^2 = 2029.79 \tag{4}$$

The sample standard deviation s is therefore

$$s = 45.05 \tag{5}$$

The 98-percent confidence interval for the population mean is given by

$$84.12 - t_{.01,31}\left(\frac{45.05}{\sqrt{32}}\right) \leq \mu \leq 84.12 + t_{.01,31}\left(\frac{45.05}{\sqrt{32}}\right) \tag{6}$$

$$84.12 - 2.46(7.96) \leq \mu \leq 84.12 + 2.46(7.96) \tag{7}$$

$$64.53 \leq \mu \leq 103.71 \tag{8}$$

It can be asserted that, if the test that produced the data of Table I is repeated many times, the sample mean of the test data will fall within the interval shown above 98 percent of the time. Going one step further, it can be stated that, 99 percent of the time, the charging current will average 103.71 microamperes or less.

The 98-percent confidence interval for the population standard deviation σ can be determined from the following:

$$\frac{31(2029.79)}{\chi^2_{.01,31}} \leq \sigma^2 \leq \frac{31(2029.79)}{\chi^2_{.99,31}} \tag{9}$$

$$\frac{62,923.49}{52.17} \leq \sigma^2 \leq \frac{62,923.49}{15.67} \tag{10}$$

$$34.73 \leq \sigma \leq 63.37 \tag{11}$$

The same parameters can be determined from the data set contained in Table II.

$$\bar{x} = \frac{30 + 40 + \ldots + 50}{45} \tag{12}$$

$$\bar{x} = 62.89 \tag{13}$$

$$s^2 = \frac{(30-62.89)^2 + (40-62.89)^2 + \ldots + (50-62.89)^2}{44} \tag{14}$$

$$s^2 = 1042.78 \tag{15}$$

$$s = 32.29 \tag{16}$$

The 98-percent confidence intervals on the population mean and standard deviation are determined as follows:

$$62.89 - t_{.01,44}\left(\frac{32.29}{\sqrt{45}}\right) \leq \mu \leq 62.89 + t_{.01,44}\left(\frac{32.29}{\sqrt{45}}\right) \tag{17}$$

$$62.89 - 2.39(4.81) \leq \mu \leq 62.89 + 2.39(4.81) \tag{18}$$

$$51.39 \leq \mu \leq 74.39 \tag{19}$$

$$\frac{44(1042.78)}{\chi^2_{.01,44}} \leq \sigma^2 \leq \frac{44(1042.78)}{\chi^2_{.99,44}} \tag{20}$$

$$\frac{45,882.32}{68.63} \leq \sigma^2 \leq \frac{45,882.32}{25.22} \tag{21}$$

$$25.86 \leq \sigma \leq 42.65 \tag{22}$$

It is of interest to examine the statistically worst case; that is, the combination of highest μ with greatest σ. Based upon the data in Table I and the calculations from those data, it can be concluded that there is a 99 percent confidence level on $\mu \leq 103.71$; in other words, if the measurements of charging current are repeated a large number of times under similar circumstances, 99 percent of the time the absolute magnitude of the average charging current will be less than or equal to 103.71 microamperes. There is also a level of confidence of 99 percent on $\sigma \leq 63.37$, using the Table I assumptions. When charging current measurements are repeated a large number of times, the square root of the sample variance--the estimate of the standard deviation--will exceed 63.37 only 1 percent of the time.

If a normal distribution for a population having these parameters is constructed, statements can be made regarding the probability of a current reading of any value, and a level of confidence can be assigned to such statements. The probability associated with statements based upon this worst-case distribution would be the product of the probabilities associated with the worst-case values of μ and σ; that is,

$$(.99) \times (.99) = .98 \tag{23}$$

Figure 6 illustrates the probability of a charging current of equal to or less than x_i versus x_i. Curve A is plotted on the

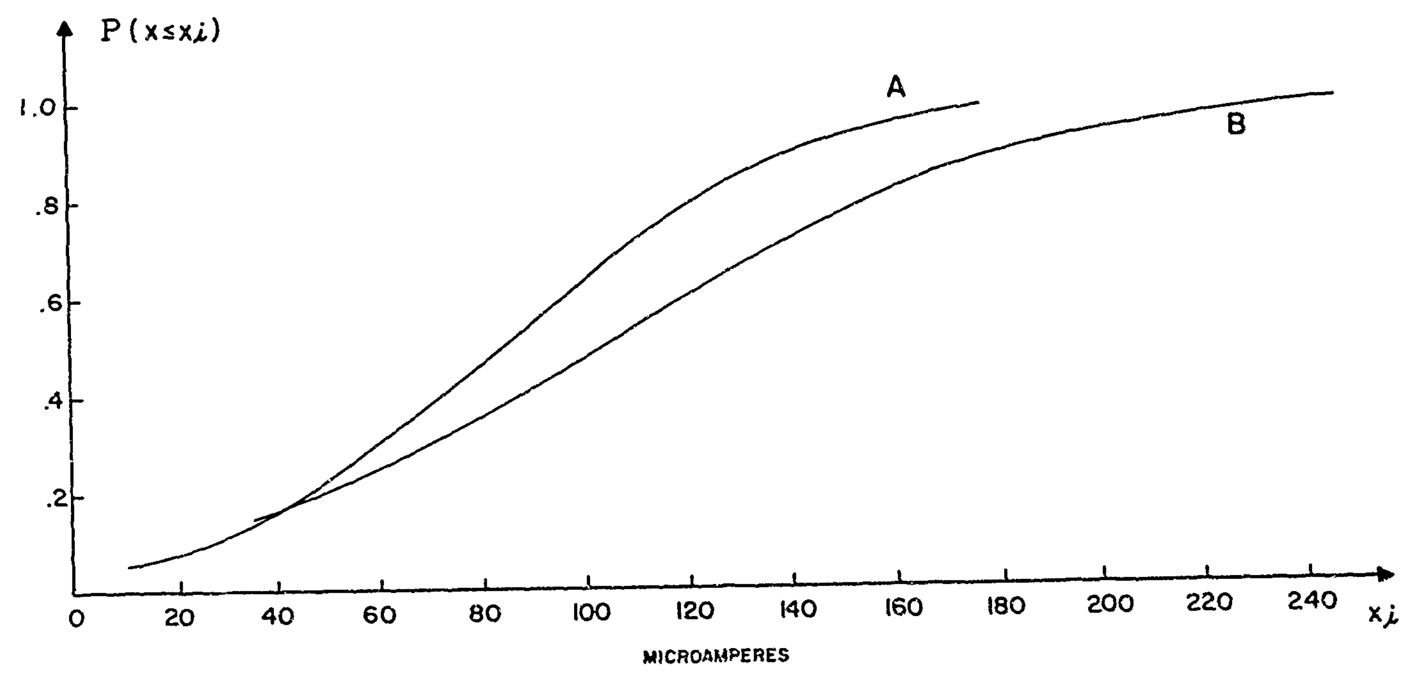

Figure 6. Cumulative Probability Distribution of Natural Charging Current Experienced by CH-54A, Based Upon Data in Table I (Curve A) and Worst-Case Calculations From Data in Table I (Curve B).

basis of the $\bar{x}$ and s of the data set in Table I. Curve B utilizes the worst-case values of μ and σ calculated from the Table I data.

Therefore, based upon Curve B in Figure 6, it can be stated with a probability or level of confidence of 98 percent that the charging rate experienced by the CH-54 helicopter in a high-charging environment such as that encountered at Yuma will be less than or equal to 200 microamperes 94 percent of the time. The CH-54 will experience charging rates of 185 microamperes or less 90 percent of the time.

This conclusion is based upon some reasonable assumptions, and, naturally, its scope is limited to dust-charging situations such as those experienced when the data were taken. In addition, the test runs which resulted in the data used for these calculations did not include the flare approach. It is possible that, had this procedure been employed, somewhat higher sample means may have been realized.

These considerations are germane to placing the foregoing statistical analysis in proper perspective, but they do not limit its usefulness. The conclusions presented are based upon the worst case, not only from a statistical point of view, but from an operational one. The flight crew expressed the opinion that such environments as were experienced at Yuma constituted the operational external hookup situation in Southeast Asia about 50 percent of the time.

The aircraft/ground capacitance measurements are presented in Table III. The zero-altitude figure represents the capacitance measured when the aircraft was sitting on a paved ramp, with the grounding wire probes, located in the vicinity of the landing gear, in contact with the ramp. In the ideal case--a perfect ground connection--this capacitance would be infinite.

It is to be noted that the data points at hovering altitudes are lower, by a factor of 2, than those recorded at Lakehurst.[2]

ESD Evaluation

The discharging capability test data recorded with the exciter/multiplier mounting fixture located at fuselage station 726 are presented in Table IV. The maximum gross discharge current for the positive multiplier was 130 microamperes, with a net discharging capability of 65 microamperes. The negative multiplier gross discharge was limited to 108 microamperes; the net discharge was 60 microamperes.

TABLE III. AIRCRAFT/GROUND CAPACITANCE MEASUREMENTS	
Altitude (feet)	Capacitance (microfarads)
0	2660
10	370
20	300
30	300
40	260

TABLE IV. DISCHARGING CAPABILITY TEST DATA

I_P (μA)	I_{DL} (μA)	I_R (μA)	Discharging Efficiency (%)
+ 12	+ 12	0	100
+ 18	+ 20	-	100
+ 30	+ 30	0	100
+ 40	+ 38	2	95
+ 50	+ 42	8	84
+ 72	+ 50	22	69
+ 80	+ 50	30	63
	+ 55	25	69
+ 100	+ 50	50	50
	+ 56	44	56
+ 120	+ 58	62	48
	+ 62	58	52
+ 130	+ 65	65	50
- 22	- 22	0	100
- 30	- 30	0	100
- 50	- 44	6	88
- 60	- 48	12	80
- 80	- 38	42	48
	- 51	29	64
- 100	- 40	60	40
- 108	- 60	48	56

Remarks: ESD Model D-15
E/M Location - Fuselage station 726
Aircraft gross weight - Approximately 29,000 pounds
I_N - Negligible
Date - June 18, 1969

When the exciters and multipliers were moved rearward to fuselage station 748, the capability of the system decreased sharply (see Table V), indicating that this location was too far back on the tail cone to provide a sufficiently strong wind force to the discharged ions. While the probe current from the positive unit reached 164 microamperes during this test run, the net discharge at that point was only 52 microamperes. The negative unit discharged a maximum net of 30 microamperes. It is apparent that the positive unit was influenced somewhat by the tail rotor, while the negative unit, shielded from the tail rotor airstream by the fuselage, was more susceptible to the ambient wind. This is indicated by the fact that drop line current readings of both 16 and 29 microamperes corresponded to a probe current of 30 microamperes. It is further indicated by discrepancies in the net discharge current readings, with a net of 30 microamperes corresponding to a gross of 96 microamperes and a net of 20 microamperes corresponding to a gross of 104 microamperes.

When the exciter/multiplier components were moved forward to fuselage station 704, the test results (see Table VI) were similar to those previously obtained at station 726. The maximum gross discharge from the positive units was 116 microamperes, with a net of 60 microamperes. From the negative side, a maximum gross of 92 microamperes was recorded, and a maximum net discharge of about 50 microamperes was achieved.

Table VII and Figures 7 and 8 show the discharging capability test results with the same ESD component location but with the aircraft about 6000 pounds heavier. This increased the downwash wind force exerted on the discharged ions and resulted in a maximum net discharge current of 80 microamperes from the positive units. The output of the negative units also increased to 65 microamperes net.

During the period when the aircraft was outfitted with the 6000-pound load, it was observed to charge at a rate of 8 to 10 microamperes, even while hovering over the ramp area. It appears quite likely that this was chiefly the result of engine rather than triboelectric charging, since there was no noticeable dust recirculation during the test and the charging current was a more-or-less constant 8 to 10 microamperes. The chart recordings (Figures 7 and 8) illustrate the magnitude of this charging current; namely, the magnitude of I_{DL} when I_P is zero.

When the D-04 multipliers were installed at fuselage station 704 on a "light" aircraft, the data recorded in Table VIII and Figure 9 were taken. Here, with less current being drawn

TABLE V. DISCHARGING CAPABILITY TEST DATA

I_P (μA)	I_{DL} (μA)	I_R (μA)	Discharging Efficiency (%)
+ 10	+ 5	5	50
+ 20	+ 12	8	60
+ 30	+ 27	3	90
+ 40	+ 34	6	85
+ 50	+ 38	12	76
+ 60	+ 38	22	64
+ 80	+ 42	38	53
+ 100	+ 43	57	43
+ 120	+ 52	68	43
+ 164	+ 52	112	32
- 20	- 10	10	50
- 30	- 16	14	53
	- 29	1	97
- 40	- 15	25	38
- 50	- 25	25	50
- 80	- 15	65	19
- 96	- 30	66	31
- 104	- 20	84	19

Remarks: ESD Model D-15
E/M Location - Fuselage station 748
Aircraft gross weight - Approximately 29,000 pounds
I_N - Negligible
Date - June 18, 1969

TABLE VI. DISCHARGING CAPABILITY TEST DATA

I_P (μA)	I_{DL} (μA)	I_R (μA)	Discharging Efficiency (%)
+ 10	+ 10	0	100
+ 16	+ 16	0	100
+ 19-20	+ 18-20	0.5	97
+ 28	+ 26	2	93
+ 37	+ 27-33	7	81
+ 50	+ 39	11	78
+ 60	+ 22	38	37
+ 80	+ 48	32	40
+ 84	+ 42-50	38	55
+ 100	+ 48	52	48
+ 116	+ 60	56	52
- 16	- 13-16	1.5	91
- 24	- 24	0	100
- 30	- 16-30	7	77
- 40	- 25-32	11.5	71
- 50	- 30-36	17	66
	- 35-41	12	76
- 60	- 40	20	67
- 80	- 48	32	60
	- 50	30	63
- 92	- 40-48	48	48

Remarks: ESD Model D-15
E/M Location - Fuselage station 704
Aircraft gross weight - Approximately 29,000 pounds
I_N - Negligible
Date - June 19, 1969

TABLE VII. DISCHARGING CAPABILITY TEST DATA

I_P (μA)	I_{DL} (μA)	$I_{DL}-I_N$ (μA)	I_R (μA)	Discharging Efficiency (%)
+ 30	+ 32-35	+ 22-25	6.5	78
	+ 40-43	+ 30-33	-	100
+ 40	+ 38	+ 28	12	70
	+ 46	+ 36	4	90
+ 80	+ 62	+ 52	28	65
	+ 75-80	+ 65-70	12.5	84
+ 130	+ 72	+ 62	68	48
	+ 90	+ 80	50	62
- 30	- 26	- 36	-	100
- 50	- 36-40	- 46-50	2	96
	- 38-40	- 48-50	1	98
- 80	- 48	- 58	22	73
	- 50	- 60	20	75
- 102	- 50-55	- 60-65	39.5	61
	- 55	- 65	37	64

Remarks: ESD Model D-15
E/M Location - Fuselage station 704
Aircraft gross weight - Approximately 35,000 pounds (including 6000-pound load)
I_N - 10 μA (positive)
Date - June 20, 1969

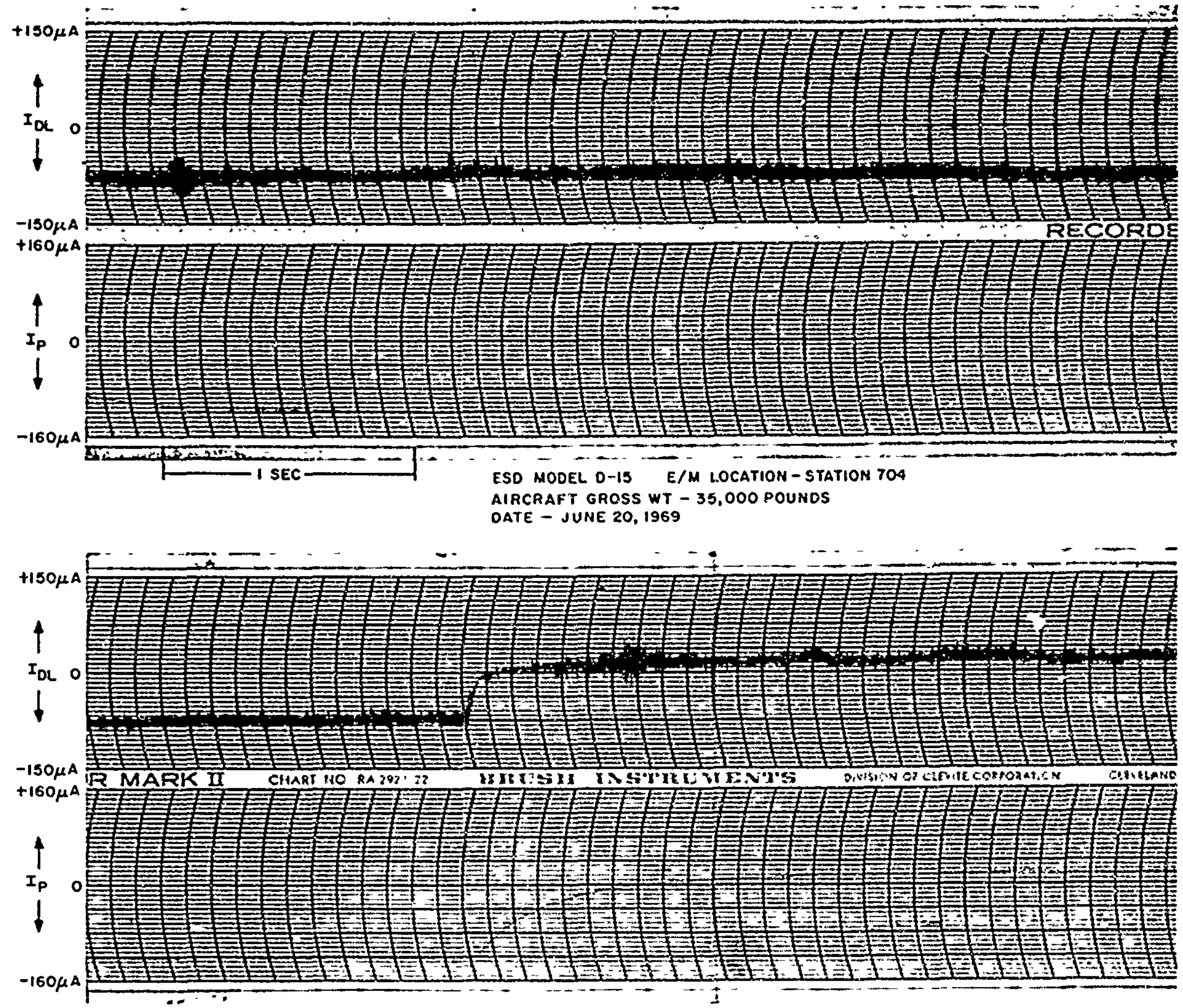

Figure 7. Discharging Capability Test Chart Recording Showing Relative Magnitudes of Drop Line and Probe Currents.

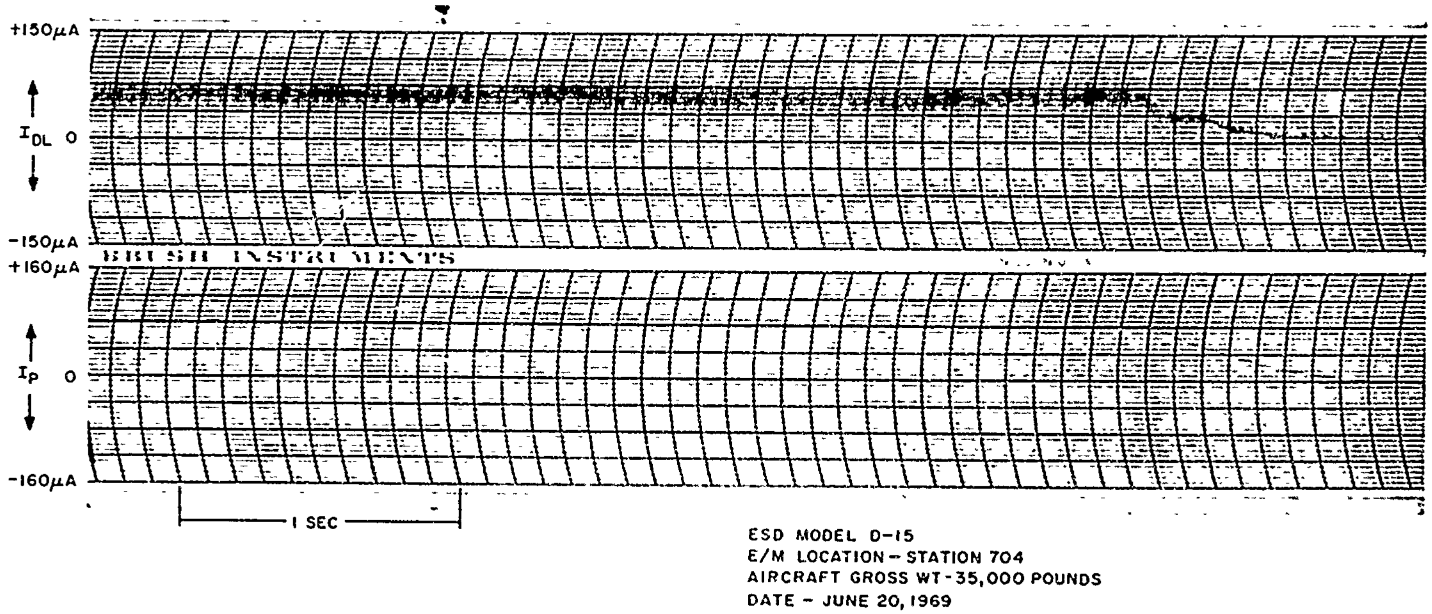

Figure 8. Discharging Capability Test Chart Recording Showing Relative Magnitudes of Drop Line and Probe Currents.

TABLE VIII. DISCHARGING CAPABILITY TEST DATA			
I_P (μA)	I_{DL} (μA)	I_R (μA)	Discharging Efficiency (%)
+ 10	+ 10	0	100
+ 20	+ 20	0	100
+ 30	+ 30	0	100
+ 40	+ 36	4	90
+ 50	+ 40	10	80
- 10	- 10	0	100
- 20	- 20	0	100
- 30	- 28	2	93
- 40	- 36	4	90
- 50	- 40	10	80

Remarks: ESD Model D-04
E/M Location - Fuselage station 704
Aircraft gross weight - Approximately 29,000 pounds
I_N - Negligible
Date - June 20, 1969

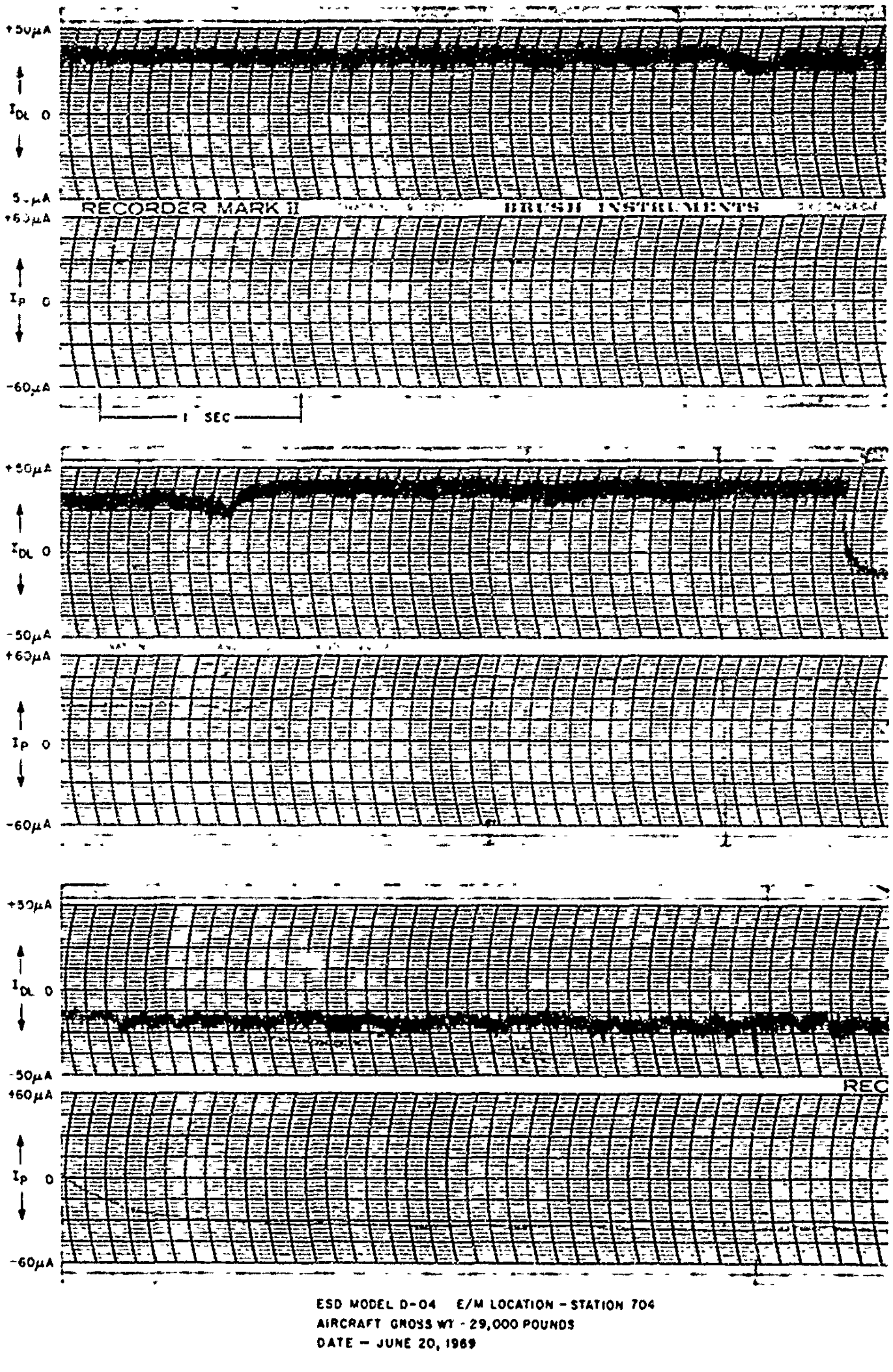

Figure 9. Discharging Capability Test Chart Recording Showing Relative Magnitudes of Drop Line and Probe Currents.

through the interconnecting lines, the positive and negative units performed in much the same manner, each being capable of discharging a net of 40 microamperes from a gross of 50.

It is to be noted that the probe current trace lags the drop line current trace in Figures 7 through 9. The probe current actually leads the drop line current somewhat, because of the procedure used; namely, adjusting the probe current as the independent variable. However, the presence of a long-time-constant circuit in the ESD probe current line results in the apparent contradiction in these chart records. This circuit does not affect the steady-state probe current recordings.

TEST PROBLEMS

Recorder

During the first day of data-taking, problems were experienced with the Clevite-Brush Mark II recorder unit. The unit was equipped with an interlock to prevent pen recording when the access door to the chart section was open. During the flight test, the vibration experienced by the recorder caused a failure in this interlock, and it was impossible during the first day of testing to activate the pen and achieve a chart record. This failure in the recorder unit was repaired, and, from the second day of flight testing to the completion, the problem did not appear again.

Meters

After the recorder malfunction had been eliminated, early set-up problems were encountered with the Keithley Model 610 Electrometer Voltmeter (EVM). These became apparent in a noncorrelation between current readings on the EVM and those on the instrumentation test panel. At that time, the EVM was removed from the test setup and the panel meters were wired in series with the test set meter as a check. Early readings indicated a fairly close correlation between the panel meters and the meter on the test set. However, there was no correlation at higher ranges; that is to say, the test set readings were approximately half those on the panel meters.

It appears most reasonable to accept the panel meter readings for three reasons:

1. Good correlation was achieved between the four panel meters of different ranges. All four were zero-center meters, and the ranges were 25, 50, 100, and 500 microamperes.

2. The panel meters were set up for this function. The test set meter was not checked out for this function.

3. A good correlation was later achieved between the EVM and the panel meters. The EVM problems evidently resulted from a faulty first installation; when the installation was torn down and the EVM was checked against the panel meters, the correlation was good, and the discrepancy did not repeat itself in subsequent flight tests.

Voltage Measurements

The high-voltage divider--designed to provide a signal equal to one-tenth of the aircraft-to-ground voltage as an input to the EVM--was misplaced by a commercial airline enroute to Yuma. However, the loss of this device did not seriously reduce the value of the data obtained.

On the basis of the data obtained thus far, it can definitely be asserted that the CH-54A accumulates an extremely high charge. However, without the aircraft voltage measurements, it is not possible to say exactly how high. The stored energy is on the order of several joules, but it is not possible to say for certain whether it is 3 joules or 10. In any event, it is an academic point; the severe shock level is 100 millijoules, and the lethal level is 1 joule. The charging current measurements recorded at Yuma combined with the V/I curves developed at Lakehurst show that the lethal level was surpassed by a considerable margin.

One series of aircraft voltage measurements was attempted.

As was illustrated earlier, the aircraft charged at about 10 microamperes while hovering over the ramp and carrying the 6000-pound load. Since this level of charging current was well within the ESD operating range, it was likely that the ESD could maintain the aircraft voltage at a level below the 30-KV maximum for the EVM probe. In that case, the high-voltage divider would not be required. The ESD system was operated under these conditions, and the aircraft-to-ground potential was recorded, as shown in Figure 10.

However, the chart recording is inconclusive as to the operation of the ESD. Instead, it points out a breakdown in the test setup. The spike change in the aircraft voltage from about 11.25 to 5.5 kilovolts (see Figure 10) was apparently the result of an arc-over from the drop line to the aircraft. As noted earlier, the high-voltage drop line (RG 8/U) intended for use with the divider unit was not used when that device was lost. It was replaced by a lighter (16 AWG) line, which did not lend itself to voltage measurement.

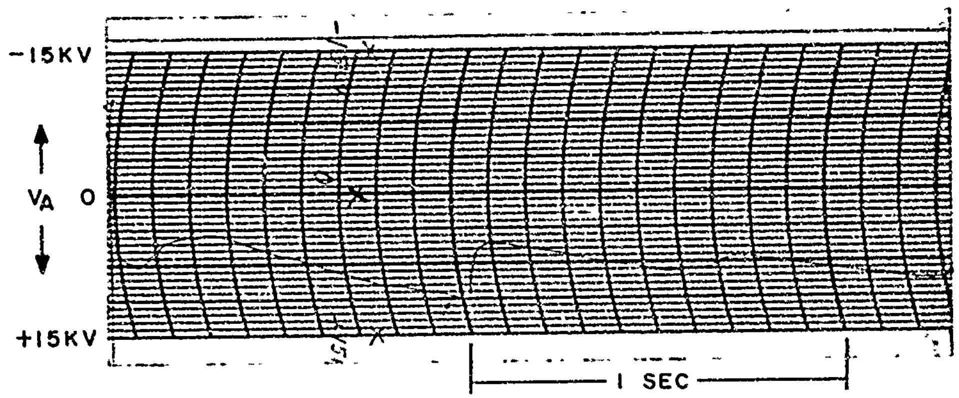

Figure 10. Aircraft Voltage Chart Recording Showing Arc-Over Resulting From Breakdown of Drop Line Insulation.

Grounding

Prior to the first flight test, it was decided that the lighter-gauge drop line with a conductive weight on the end would be used to achieve a quick ground connection and to facilitate the measurement of natural charging current during flare and transition to hover.

It became apparent, however, that the aircraft was not properly grounded by this device. When the aircraft was connected to the ground in that manner, the ESD sensor was activated and showed an external field present, which should not have been present if the aircraft and ground were at equal potentials. Subsequent to this observation, an investigation was undertaken to determine what resistance was present between the drop line and the ground. A screwdriver blade about 6 to 8 inches long was imbedded in the ground outside the hangar at Laguna Army Airfield and connected electrically to the electrometer voltmeter using the impedance range. The other terminal of the impedance meter was connected to the drop line with weight. The weight was dropped on the ground, and the impedance between the buried screwdriver blade and the drop line with weight was determined to be on the order of 2.5×10^9 ohms. The weight was then scraped on the ground, and the impedance was observed to drop by about an order of magnitude. Then, two screwdriver blades were buried in the ground about 8 to 12 feet apart; the impedance between them was 400 kilohms. The underlying theory behind these measurements is that, if there is a good ground and two conductors are in contact with the good ground and separated by some distance laterally on the surface of the ground, a low impedance will be measured between these conductors. Apparently the two screwdrivers during the last part of this test were making a considerably better ground than was the weight under any of the test conditions prior to that.

As a result of these tests, it was decided to use a grounded stake (a stainless steel rod approximately 4 feet in length, which was driven into the ground in the dust area to within approximately a foot of its full length) and to wet the ground in the area of this stake. The ground connection, however, was still apparently not good enough.

In the future, it is recommended that the test site ground be measured in a similar way as with the screwdrivers, but using grounded stakes in place of screwdrivers.

This series of impedance measurements has some important implications in terms of the feasibility of the direct grounding method for achieving safe external cargo hookup operating conditions. There is apparently a considerable resistance, R_1 (see Figure 11), between any two points on the surface ground and also between any point on the surface ground and good ground, R_2 or R_3. If an aircraft is electrostatically charged, it has a potential with respect to good ground; and, if a drop line is connected from any surface ground point A to the aircraft, the potential is equalized between point A and the aircraft. However, no charge equalization occurs between a man standing at point B and the aircraft, because a considerable impedance exists, as much as 2.5×10^9 ohms, between the man and point A, which is at the same potential as the aircraft. Therefore, even if a good connection is maintained between the aircraft and point A, the man at point B can receive a severe shock when he comes in contact with the hook. The grounding line does not necessarily provide a direct shunt for the man at point B; it provides a shunt to point A or possibly to good ground, in which case the man is shunted by the drop line in series with either R_1 or R_3.

ESD Output

The test installation suffered from the fact that supply and signal lead lengths were excessive for the wire gauge employed. The result of this situation was appreciable line losses and subsequent reduction of system output (gross discharge current).

The test data indicate a consistently lower gross capability for the negative units than for the positive. This indicates that the input impedance of the negative exciter/multiplier components was lower than that of the positive. In a production-type installation, considerable latitude in this area is permissible, because even low input impedances are quite high compared to the series line impedance. But when the line resistance is increased, normal unit-to-unit variations in parameters become quite noticeable.

The negative multiplier is always a more efficient discharger, because a higher voltage (and, thereby, a higher electrostatic field between the discharge probe and the aircraft) is required to support a given positive current than is required for a negative current of the same magnitude.

This phenomenon tended to offset the effects of the low input impedance of the negative exciter/multiplier components. As a result, the negative discharge capability of the ESD as measured at Yuma was not much less than that of the positive.

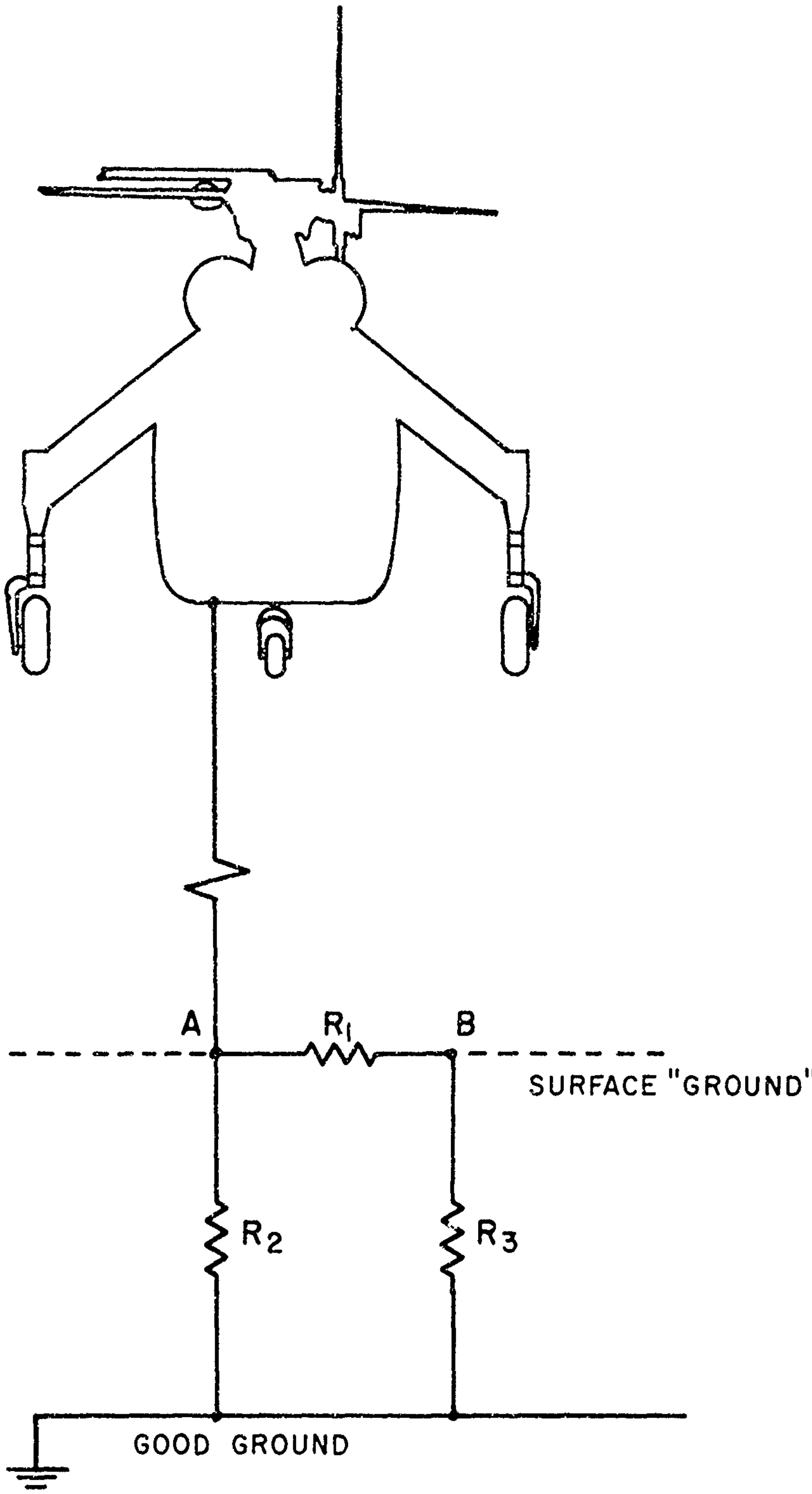

Figure 11. Surface-Point-to-Surface-Point and Surface-Point-to-Good-Ground Impedances Existing in Dusty Terrain.

CONCLUSIONS

Analysis of the results of the Yuma flight test program, in the light of earlier test information and the present state of the art, leads to the following conclusions:

1. In dusty environments, charging currents on the CH-54A can be expected to exceed 300 microamperes on occasion. Charging rates approaching 200 microamperes can be sustained for considerable periods of time.

2. Based on the data from this test program and the data from the Lakehurst tests which describe the voltage-current characteristics of the CH-54A, it is concluded that the unprotected aircraft will accumulate voltage and energy levels which are considered hazardous to personnel and equipment when operating in conditions similar to those experienced during this test program.

3. A discharger system for the CH-54A helicopter should have a minimum net discharging capacity of 200 microamperes.

4. The equipment tested does not have a 200-microampere capability.

5. With the aircraft gross weight roughly 29,000 pounds, the D-15 ESD system achieved discharge currents of approximately 130 microamperes gross and 65 microamperes net. When the gross weight of the aircraft was increased to about 35,000 pounds, the maximum net discharge current increased to approximately 80 microamperes.

6. Based on the discharging capability test data from this program and from the Lakehurst tests with the exciter/multiplier bracket located at fuselage station 644, it is concluded that the multipliers could be mounted at fuselage station 726 and as much as 3 or 4 feet forward of that point without appreciable deterioration of the discharging efficiency. This means that two multiplier units could be installed on the same side of the aircraft and separated sufficiently far to preclude interaction.

RECOMMENDATIONS

Based upon the conclusions presented on the preceding page, the following recommendations are offered to provide the most expedient means to a final solution for the CH-54 electrostatic charging problem:

1. The Dynasciences Model D-15 active ESD system should be modified to increase its net discharging capability to 200 microamperes. Specifically, the multiplier unit should be repackaged to reduce recirculation and to provide a minimum net discharging capacity of 100 microamperes; the present system should be expanded to include two sets of exciter/multiplier components of each polarity.

2. A permanent installation should be designed and fabricated for the D-15 ESD system on the CH-54A helicopter. This installation should be executed on one aircraft in the United States, so that the modified ESD system can be flight tested in a high-charging environment.

3. After obtaining satisfactory results from the first test installation, the Army should consider a retrofit program to outfit all CH-54 helicopters with ESD systems.

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APPENDIX I
DESCRIPTION OF TESTS

GENERAL

The CH-54 program involved basically two general groups of tests: the first was aimed at establishing the magnitude of the problem, and the second was designed to evaluate the performance of the active discharger systems.

Only one helicopter, CH-54A #18456, was involved in the test program at Yuma. The measurements were made while the aircraft was hovering at altitudes normally encountered during external cargo hookup operations; that is, up to 40 feet. Most of the tests were conducted with the CH-54A carrying no load, with a gross weight of approximately 29,000 pounds. Some measurements were made while the aircraft was carrying a load of about 6000 pounds, hooked up in a multipoint configuration; that is, with a gross weight of approximately 35,000 pounds.

A conductive drop line, or tether, was used to provide electrical connection between the hovering aircraft and the earth ground. The aircraft was outfitted with a bank of four microammeters of various ranges, which could be used to read the drop line current. The microammeters were situated on an instrumentation rack specially designed to fit in the CH-54 cockpit. This rack (see Figures 12 and 13) had previously been used during the Lakehurst test series.

A Clevite-Brush Mark II direct-writing two-channel recorder was installed on the top of the instrumentation rack in the helicopter, as shown in Figure 14. One recorder channel was used to record the ESD probe current analog supplied by a Dynasciences Model TS-17M Test Set, which was also located on the aircraft instrumentation rack (see Figures 12 and 13) to permit control and calibration of the ESD. The other channel was used to transcribe the output signal of a Keithley Model 610 Electrometer Voltmeter (EVM), which was used to measure the natural charging current and the aircraft voltage. This was installed next to the recorder on the top of the instrumentation rack, as pictured in Figure 14.

The polarity convention for charging current established during earlier work has been maintained in this report. Therefore, a positive charging current is one which results in a positively charged helicopter.[4] In keeping with this convention,

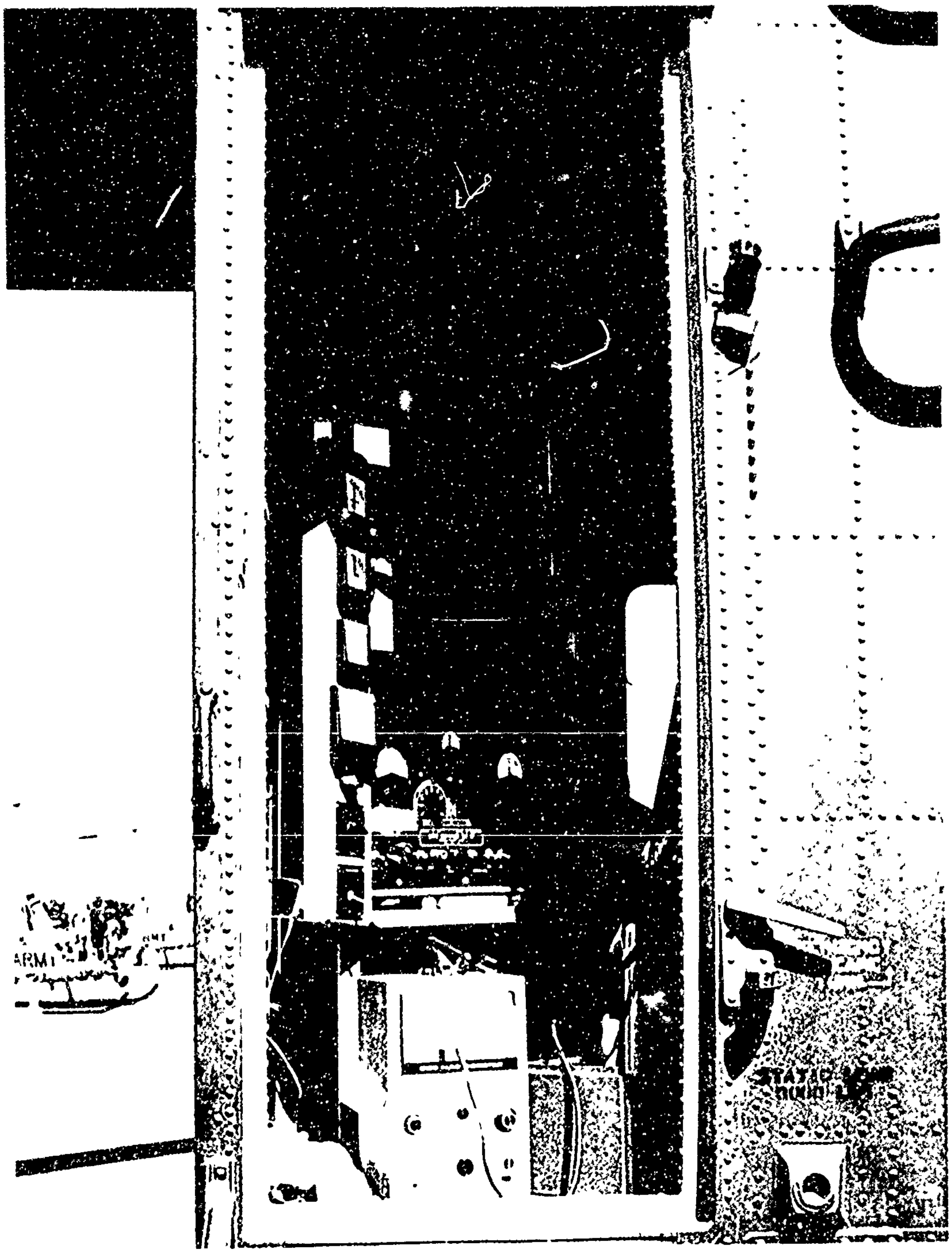

Figure 12. Test Equipment: Instrumentation Rack, Electrometer Voltmeter, and Recorder Prior to Installation in CH-54A Cockpit.

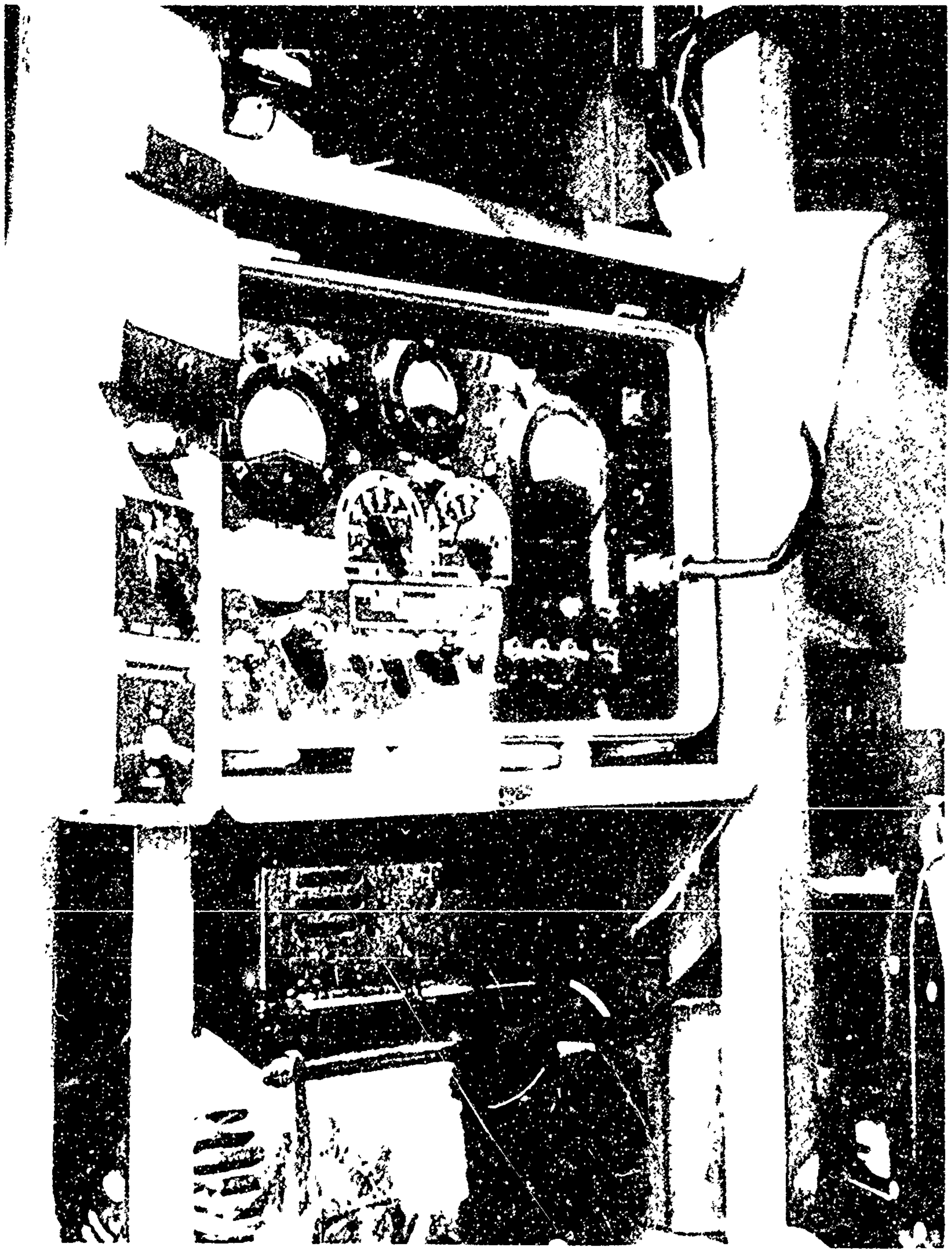

Figure 13. Test Equipment: Detail of Lower Section of Instrumentation Rack Showing Static Discharger Test Set (TS-17M), AC-to-DC Converter, and Part of Microammeter Bank.

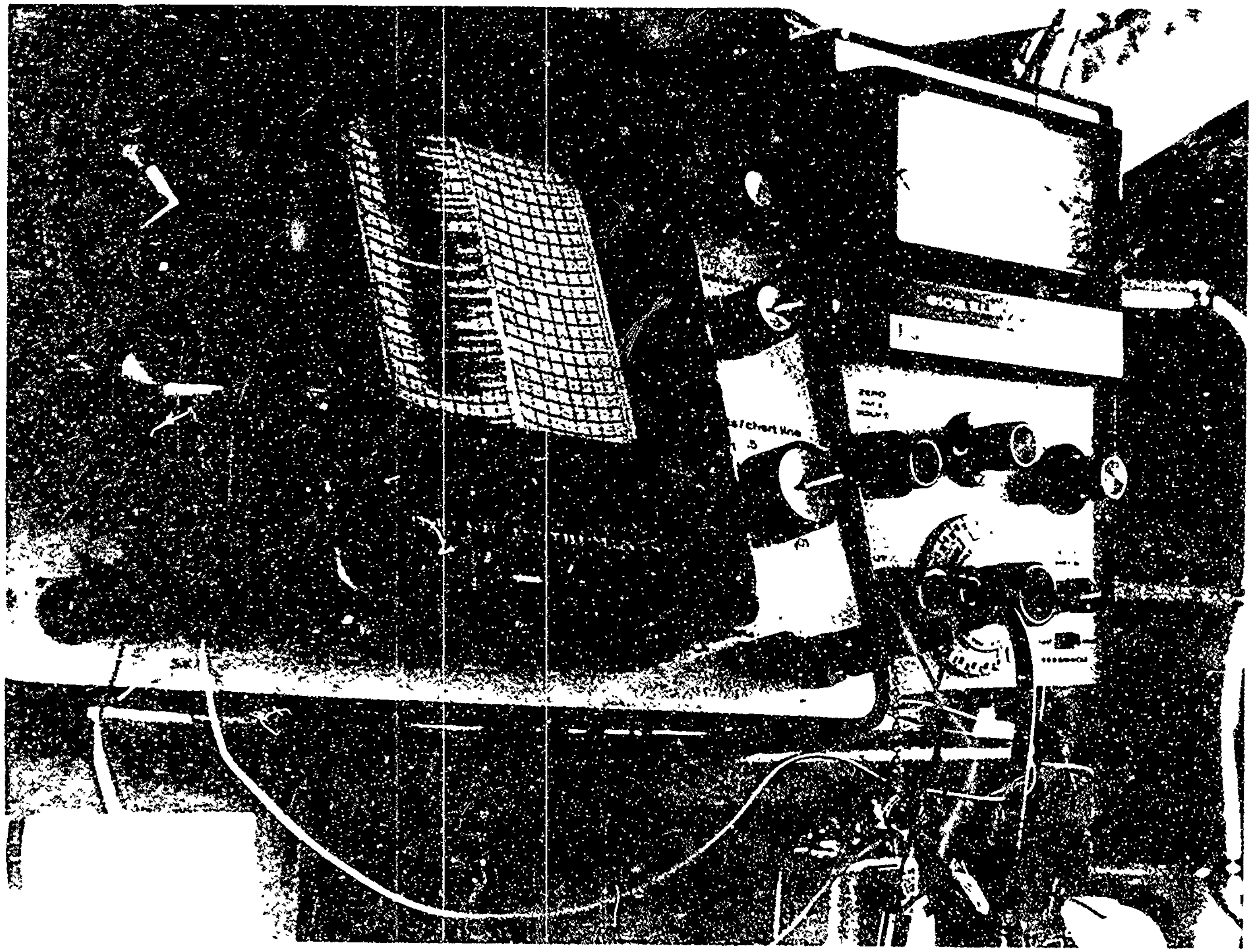

Figure 14. Test Equipment: Detail of Recorder and Electrometer Voltmeter Mounted on Top of Instrumentation Rack.

the polarities of parameters directly measured or inferred from direct measurements during this test program are illustrated in Figure 15. The arrows indicate the direction of positive current flow. It is seen that a positive natural charging current I_N causes a positive random discharge current I_{DR}. When the ESD system is operated, a positive natural charging current results in a positive probe current I_P. In later sections of this report which detail specific test procedures, it will be shown that a positive drop line current I_{DL} corresponds to either a positive ESD probe current or a negative natural charging current.

Tie in to the aircraft electrical system was accomplished in the forward section of the cockpit at the panels marked AC PRI BUS and DC PRI BUS (see Figure 16). A circuit breaker rated at 2 amperes was used in the 115V/400Hz line to the ESD test set. A 5-ampere breaker was installed in the 28-VDC line to the test set. Breakers rated at 7 and 35 amperes were put in the inverter control and main lines, respectively.

The ESD sensor was installed in a bracket located in the area of the landing lights on the access door at the front of the cockpit (see Figures 17 through 23). The ESD exciters and multipliers were installed in the same specially designed mounting bracket used at Lakehurst and strapped on to the tail cone near fuselage station 726, as shown in Figures 17 and 24 through 30. During the test, the bracket with the exciters and multipliers was moved to station 748 and finally to station 704 (Figures 31 through 33). The positive exciter and multiplier were always situated on the left (tail rotor) side of the tail cone.

The aircraft hovering altitude was estimated by the flight engineer.

AIRCRAFT/GROUND CAPACITANCE MEASUREMENT

The aircraft and the earth form the two plates of a capacitor, the value of which is a measure of the ability of the aircraft to store charge. The value of the aircraft/ground capacitance is primarily a function of the hovering altitude of the aircraft.

The capacitance of any capacitor is a function of the distance between the plates. This is explained by considering an idealized parallel-plate capacitor (see Figure 34). The capacitor is formed by two parallel plates of identical area

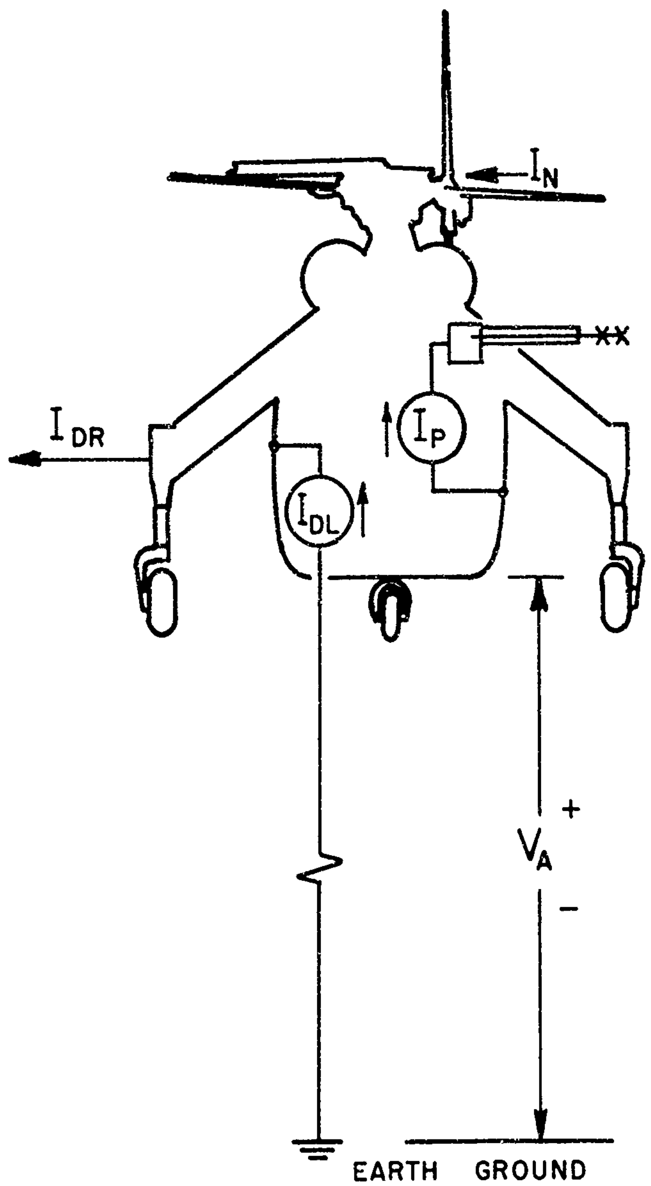

Figure 15. Current and Voltage Polarity Convention.

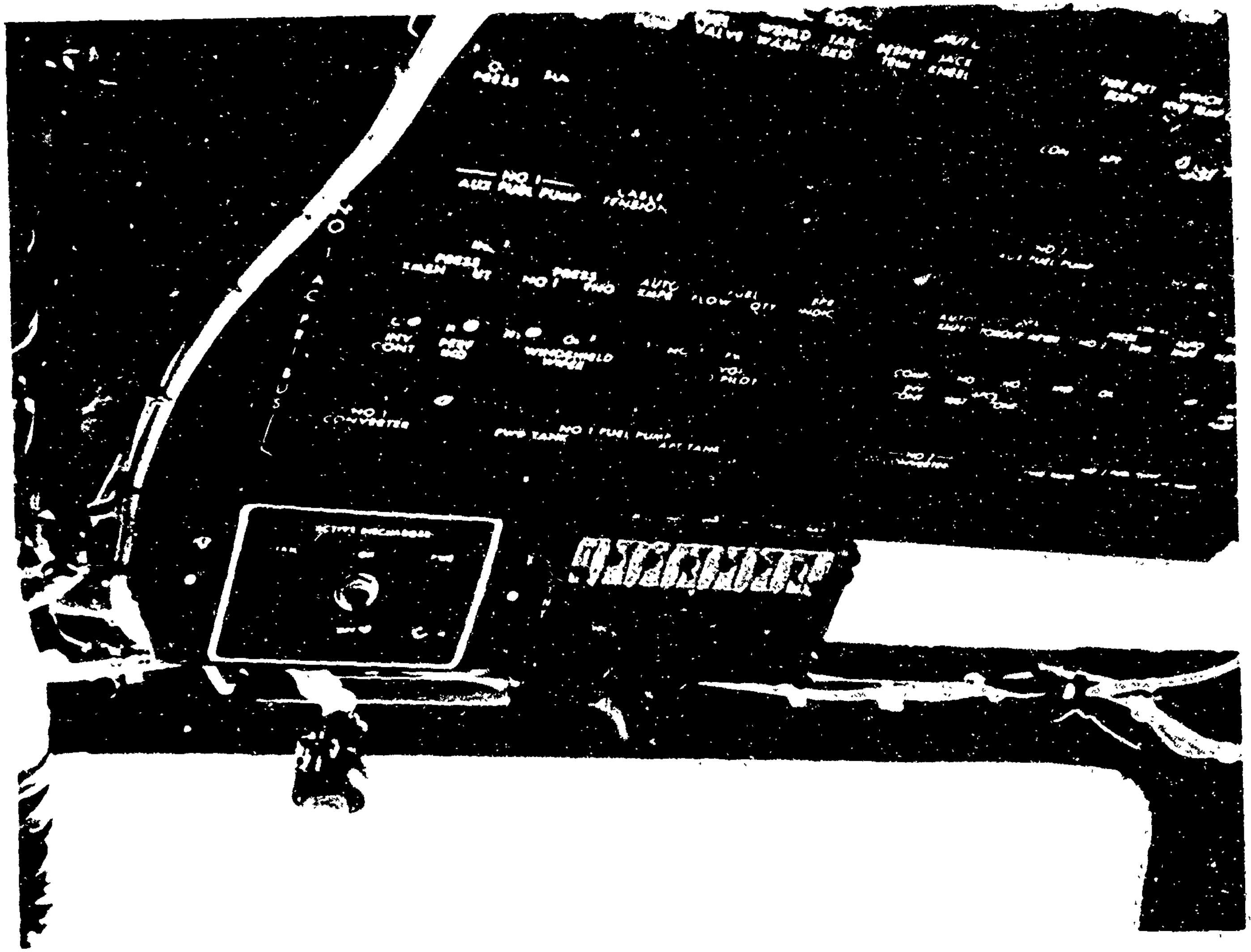

Figure 16. Tie-in With CH-54A Test Aircraft Electrical System.

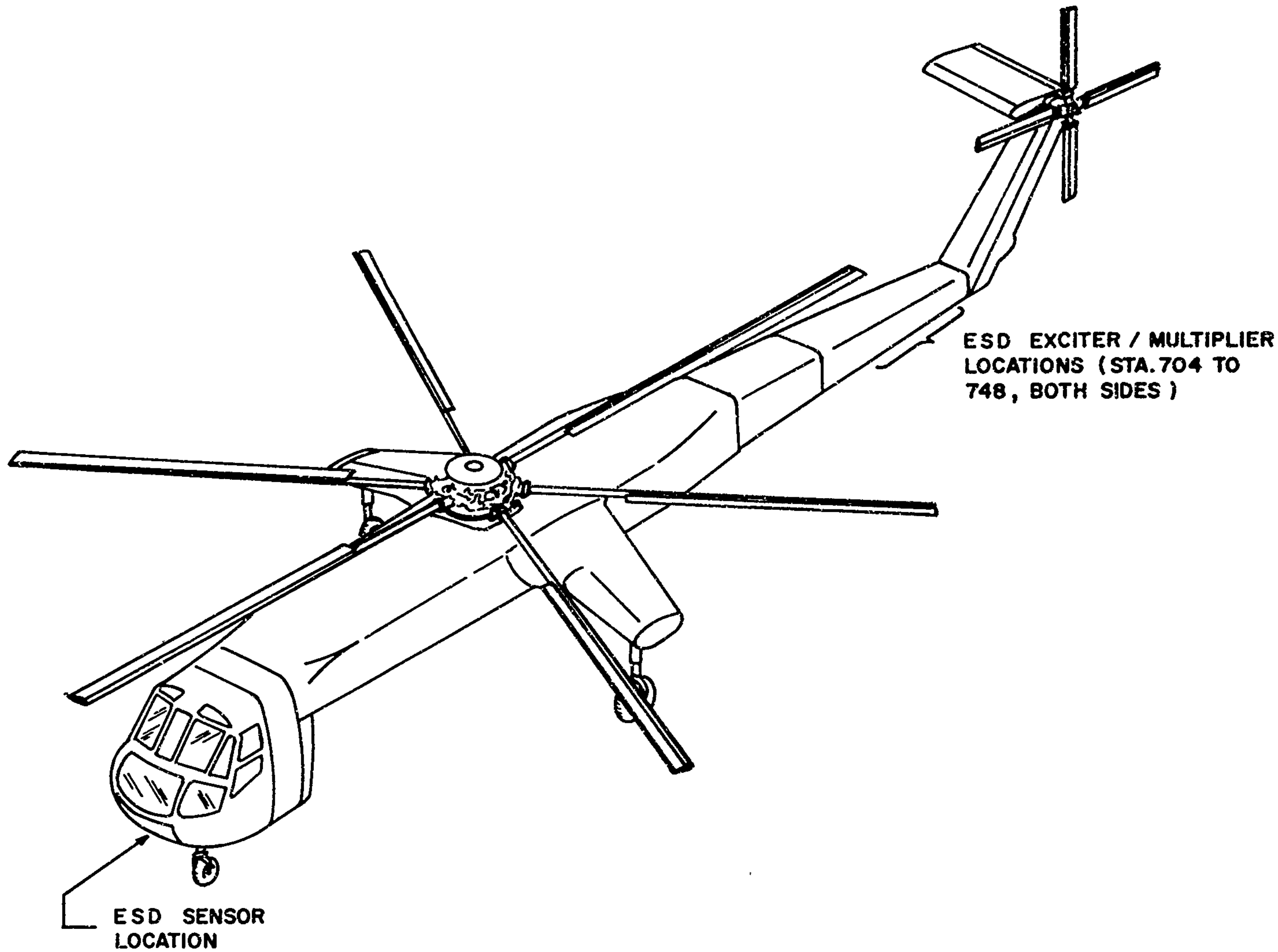

Figure 17. ESD Component Locations on CH-54A Test Aircraft.

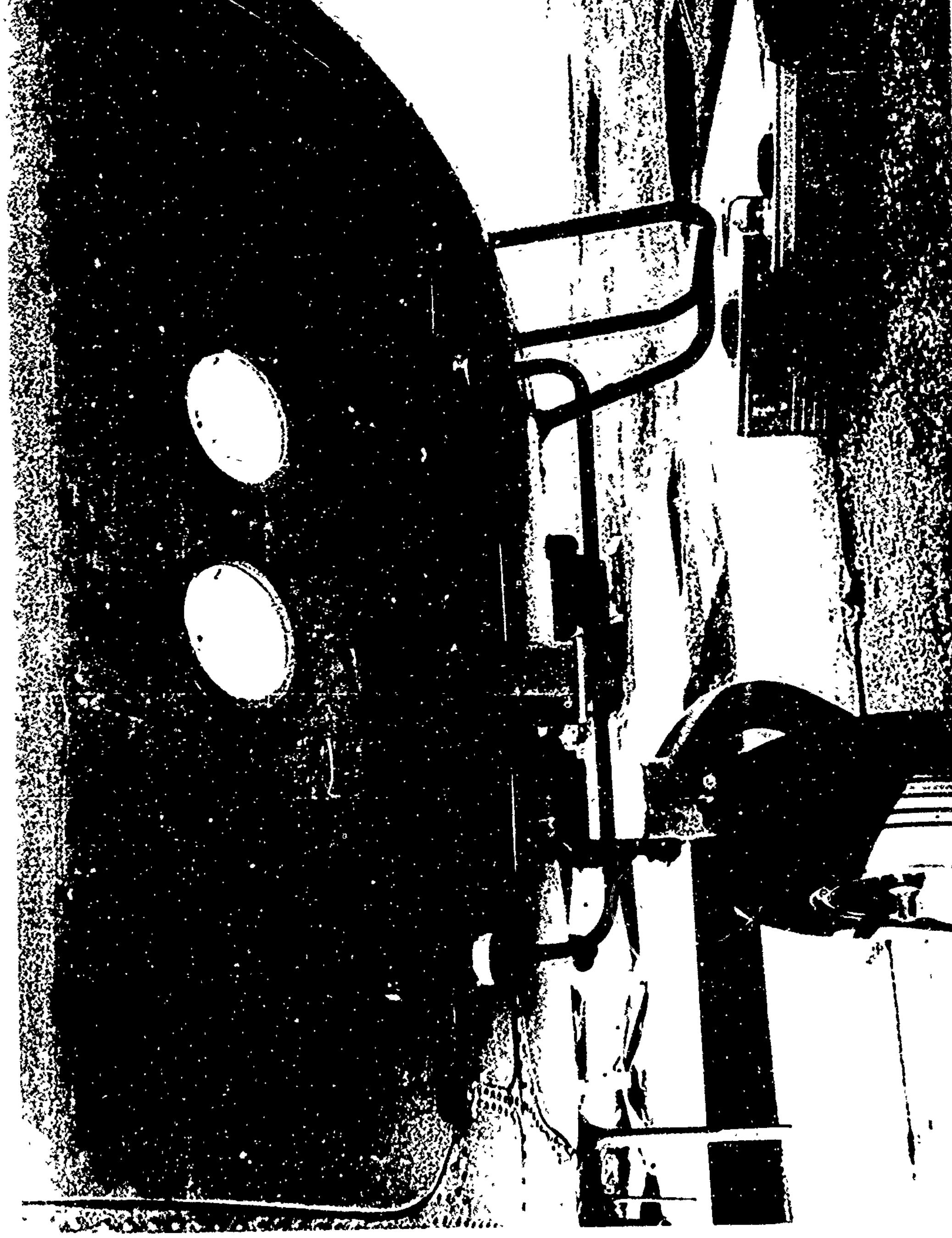

Figure 18. Mounting Location for ESD Sensor Unit.

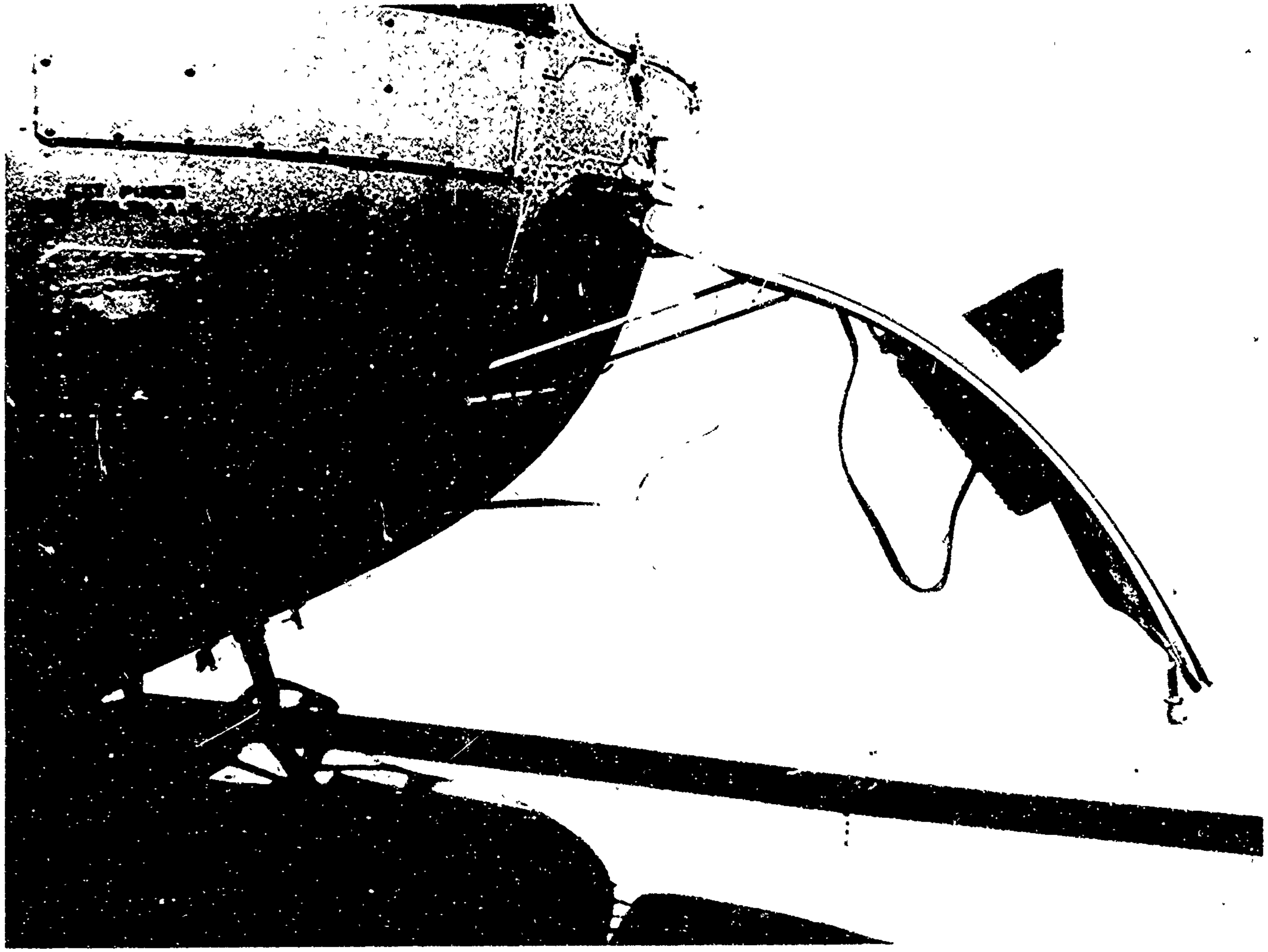

Figure 19. Preparation for Mounting of ESD Sensor Unit in Place of Landing Lights.

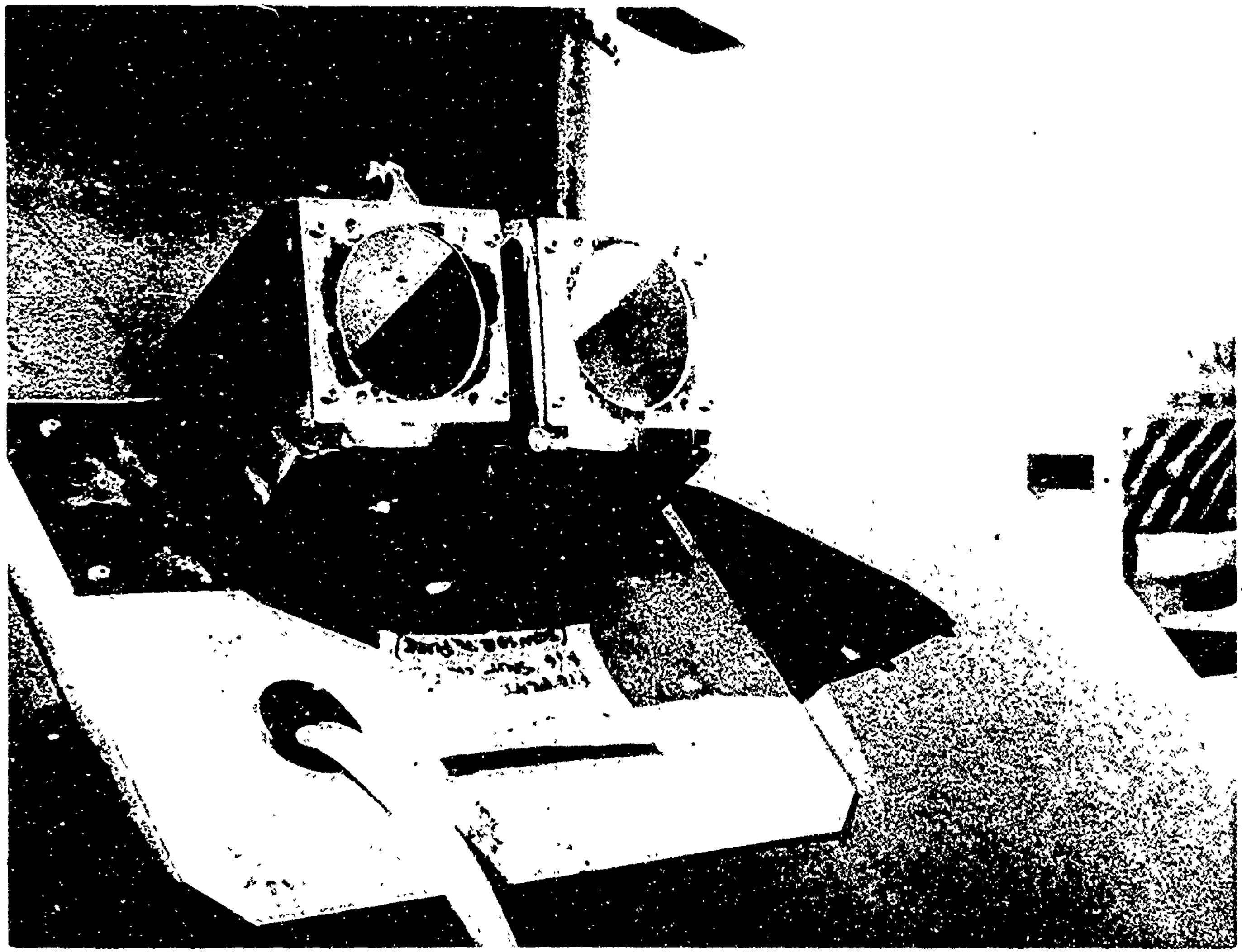

Figure 20. Detail of Mounting Bracket for ESD Sensor Unit Located in Place of Landing Lights (Front View).

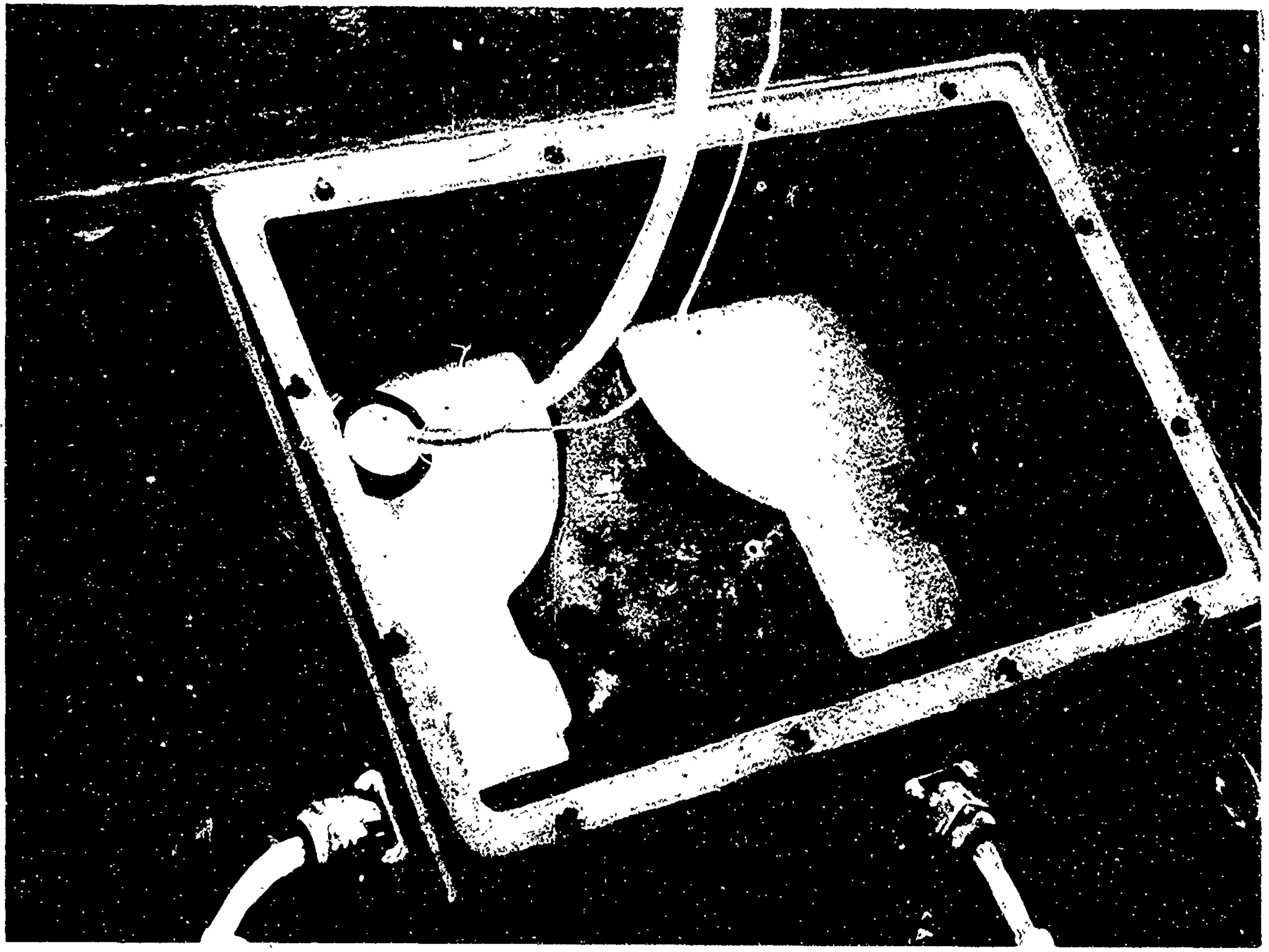

Figure 21. Detail of Mounting Bracket for ESD Sensor Unit Located in Place of Landing Lights (Rear View).

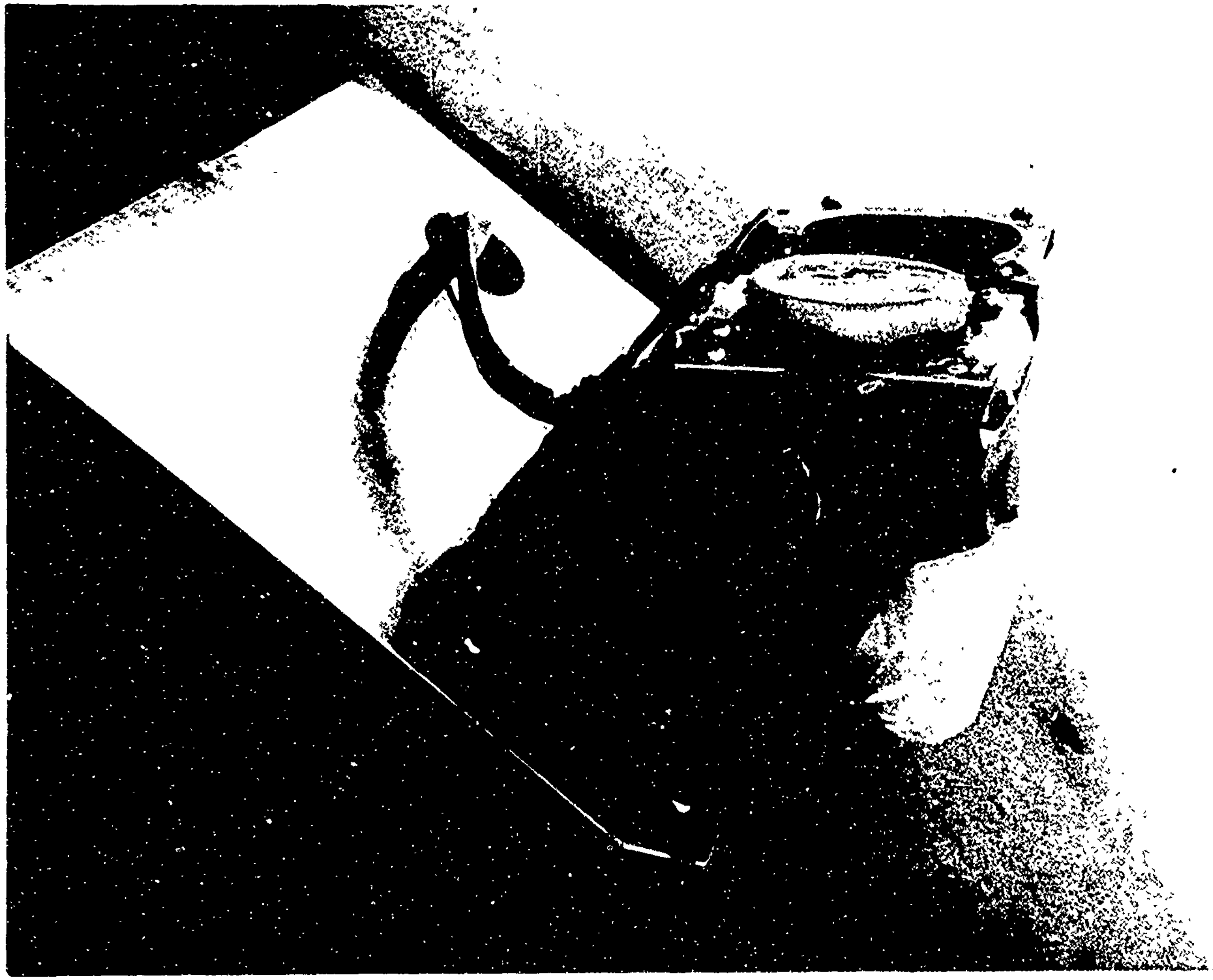

Figure 22. ESD Sensor Unit Test Installation (Side View).

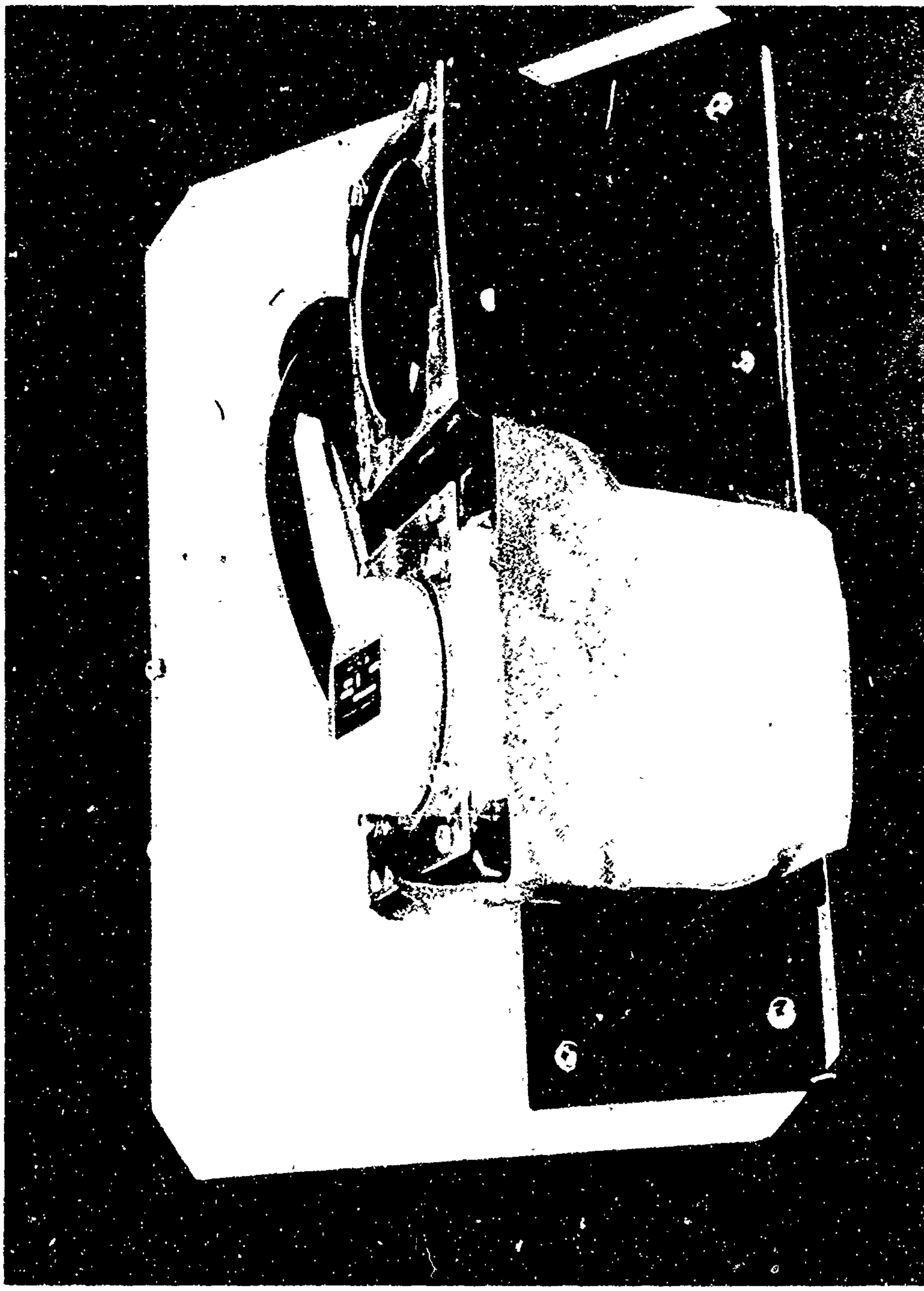

Figure 23. ESD Sensor Unit Test Installation (Front View).

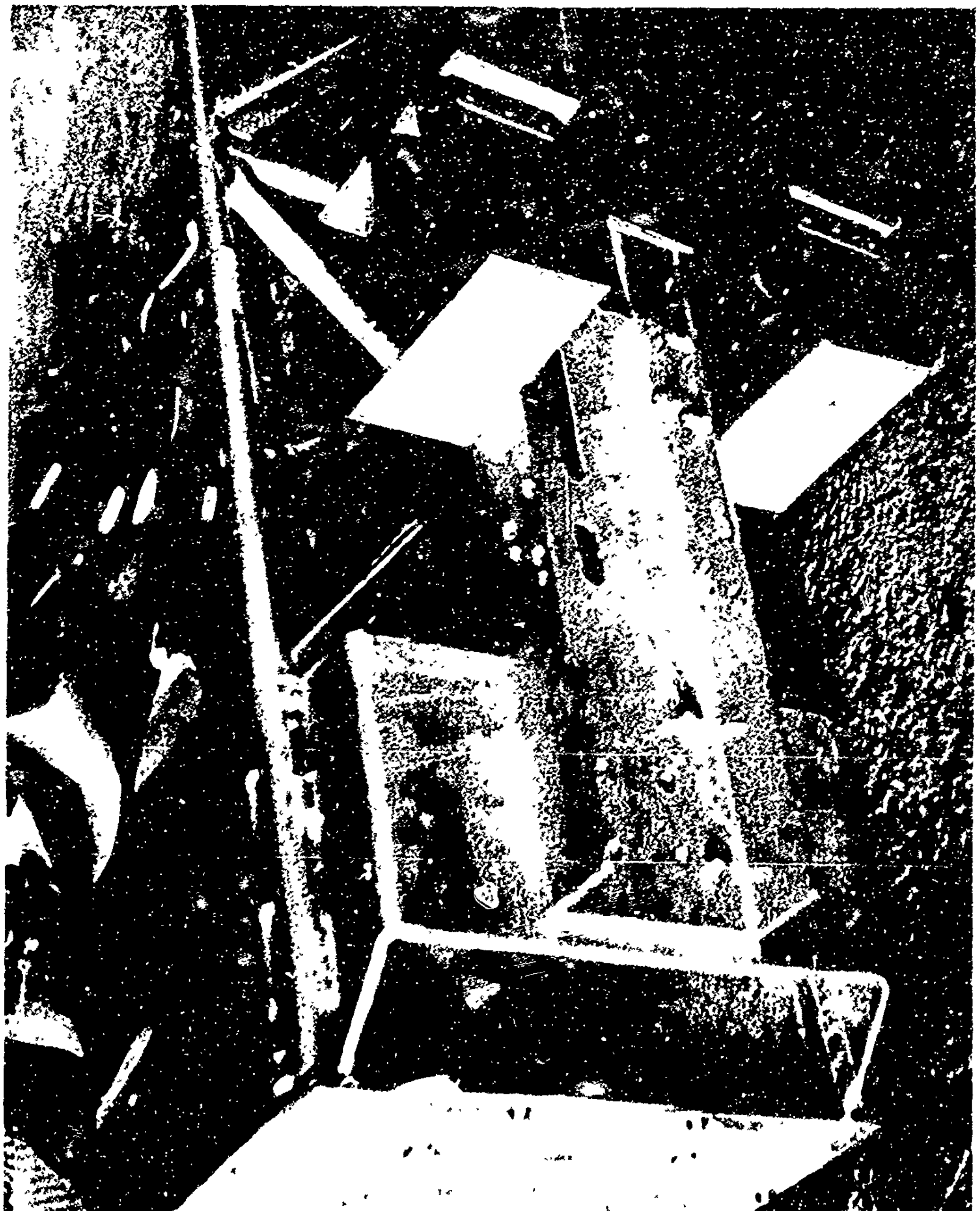

Figure 24. Detail of Section of Exciter/Multiplier Mounting Fixture With Exciter Unit Installed.

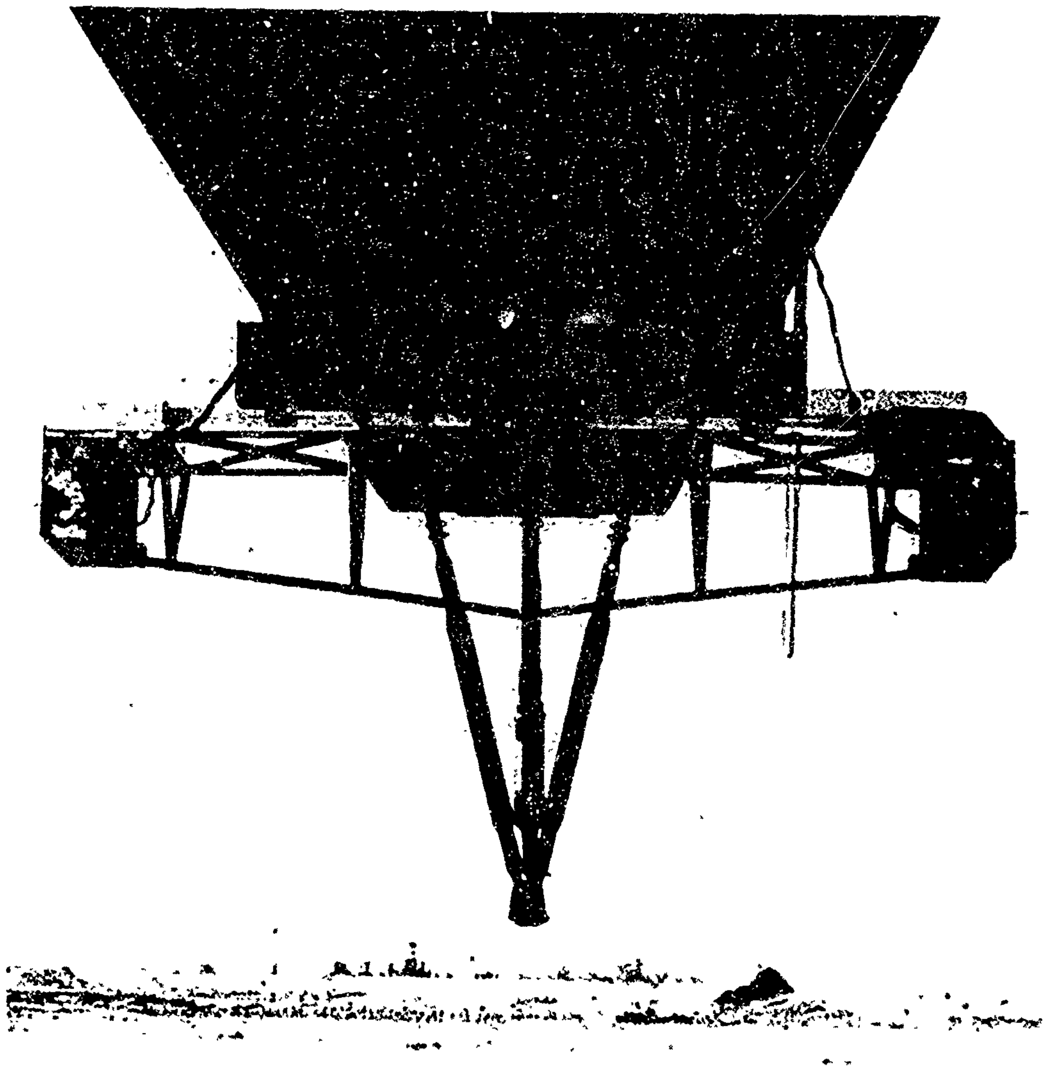

Figure 25. Exciter/Multiplier Mounting Fixture, With Exciter Units Installed, Situated on Test Aircraft at Fuselage Station 726.

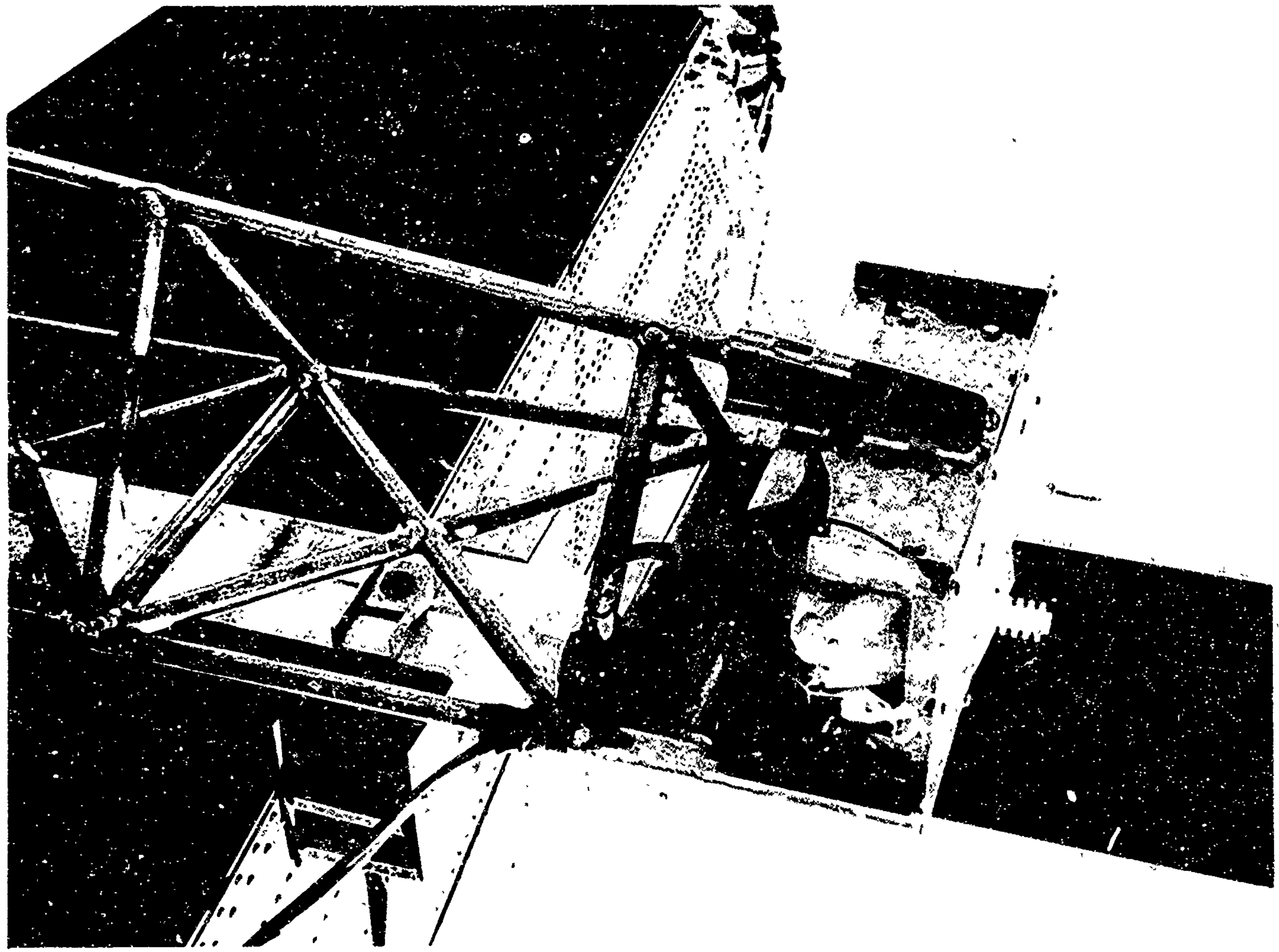

Figure 26. Detail of Exciter/Multiplier Mounting Fixture Installation and Wiring Harness Tie-in to Negative Exciter Unit (Fuselage Station 726).

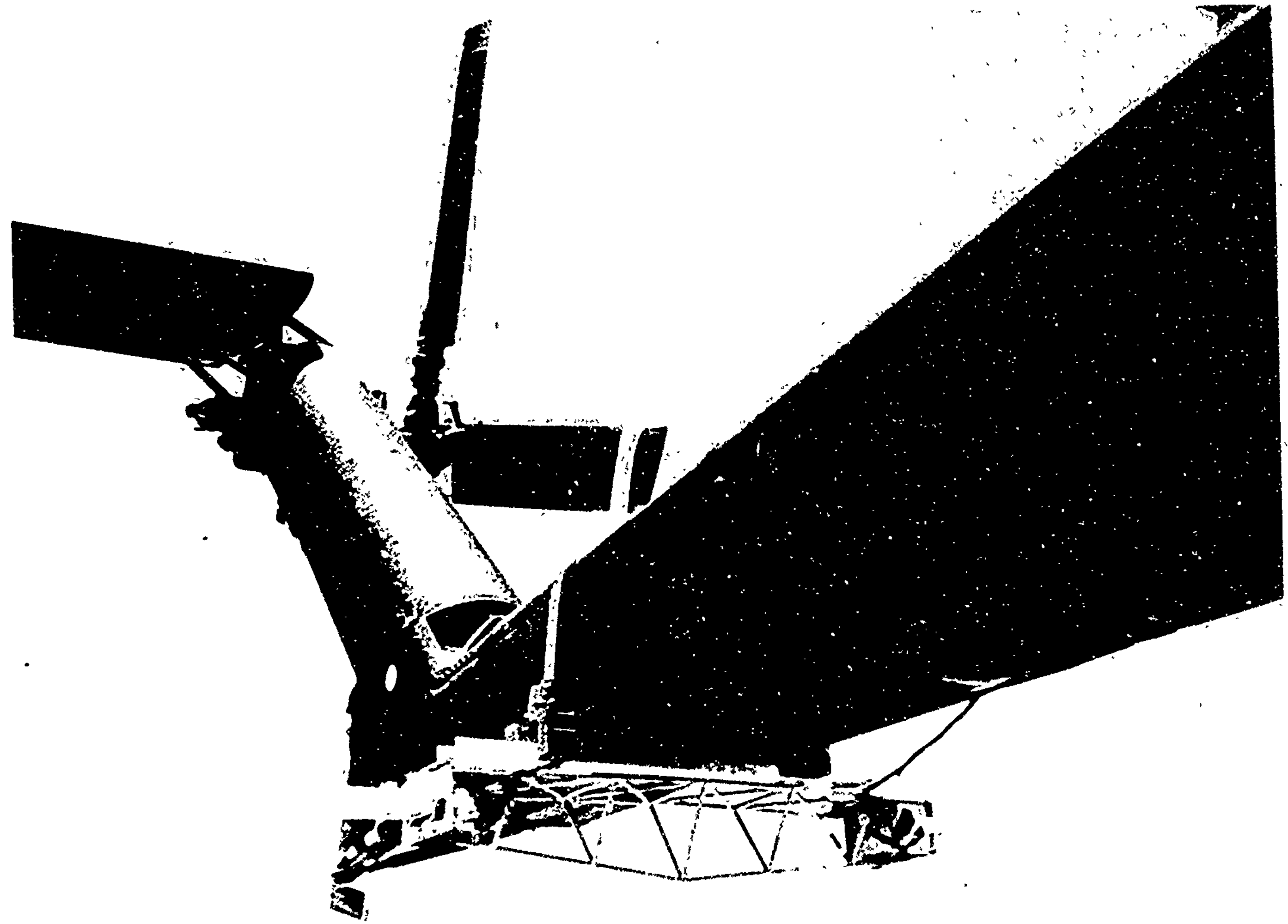

Figure 27. Completed Exciter/Multiplier Test Installation at Fuselage Station 726.

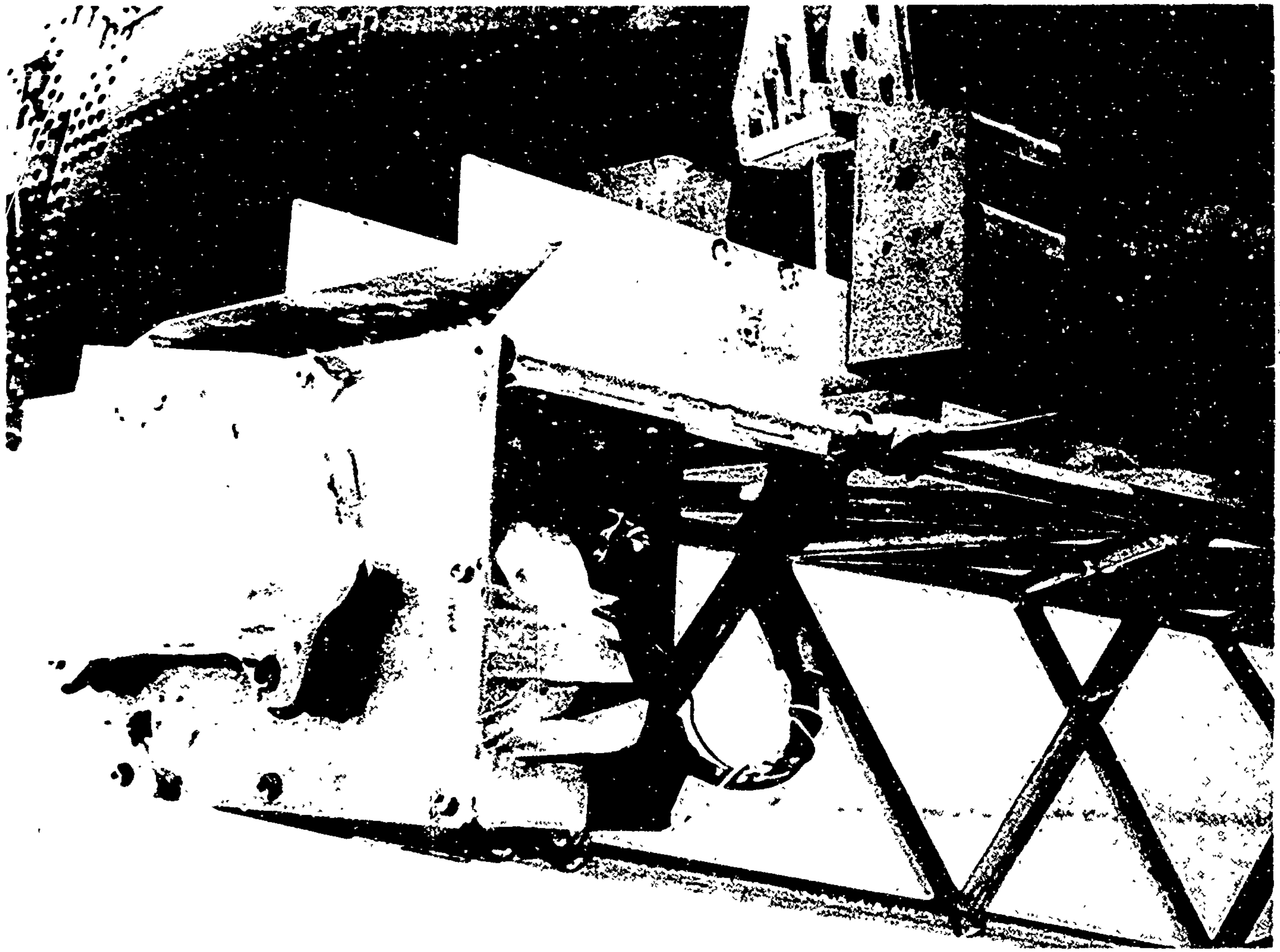

Figure 28. Detail of Exciter/Multiplier Mounting Fixture Installation, Negative Multiplier and Gasket Installation, and Wiring Harness Tie-in to Negative Exciter Unit (Fuselage Station 726).

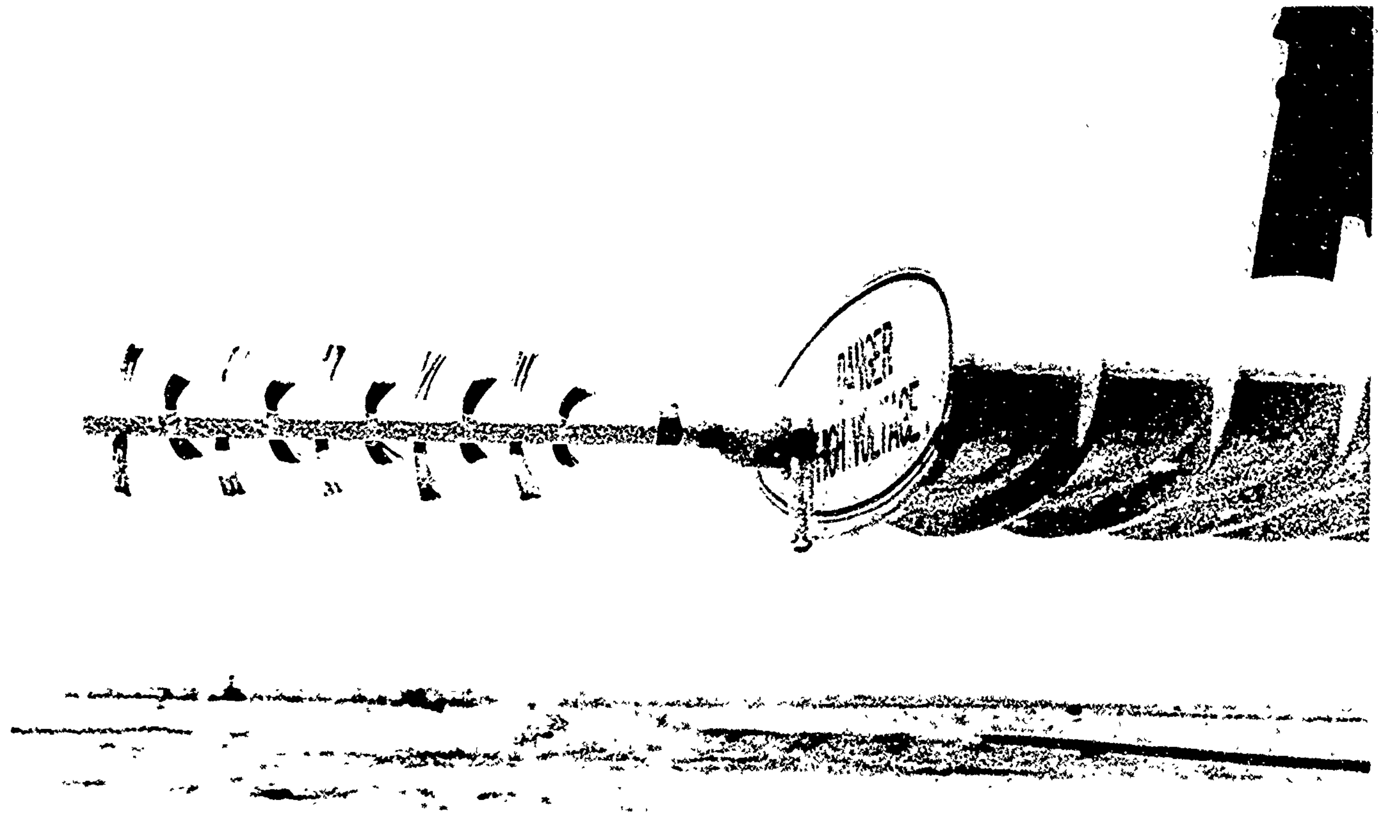

Figure 29. Detail of Outboard End of Negative Multiplier Unit Showing Discharge Probe (Station 726).

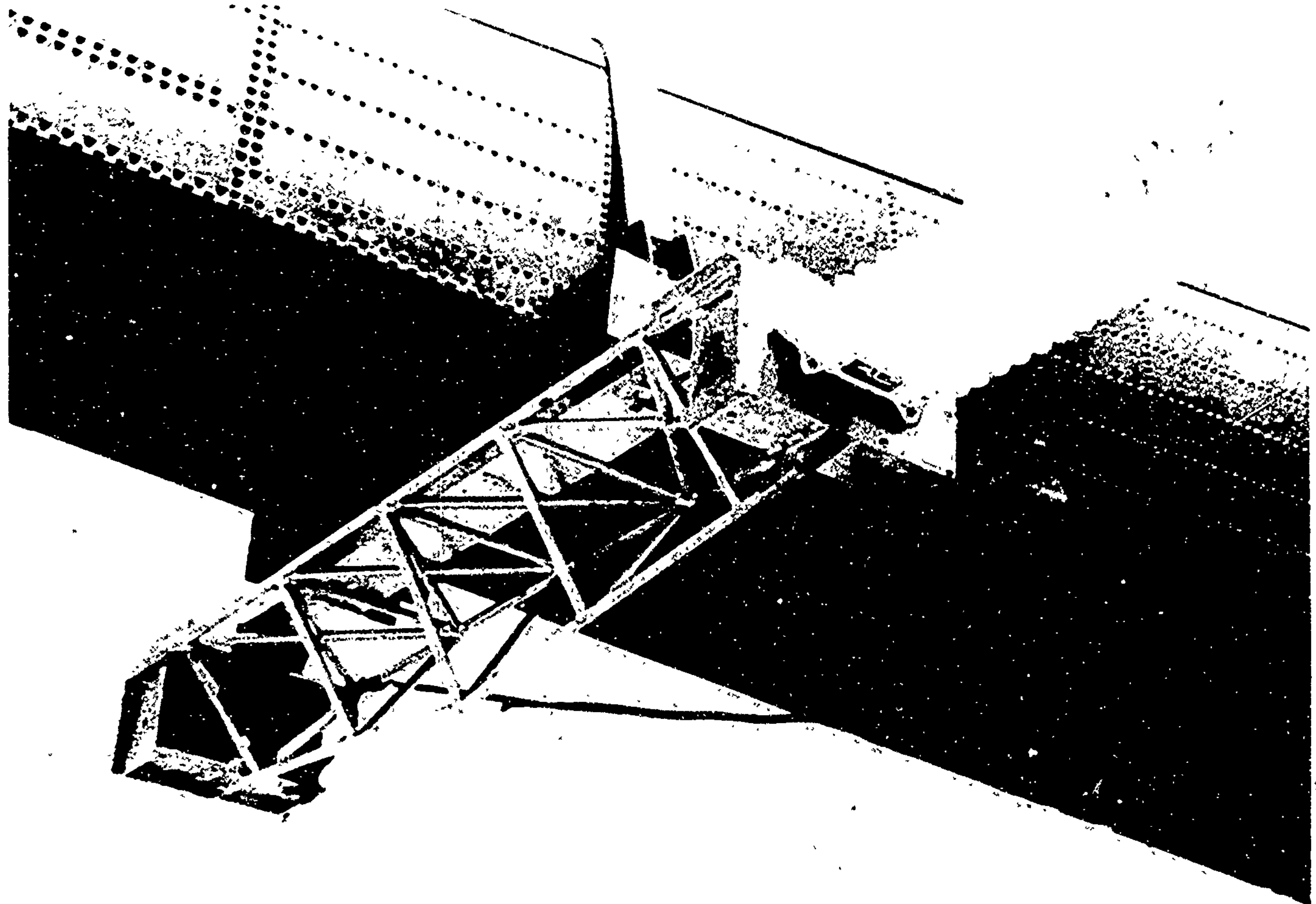

Figure 30. Completed Exciter/Multiplier Test Installation at Fuselage Station 726.

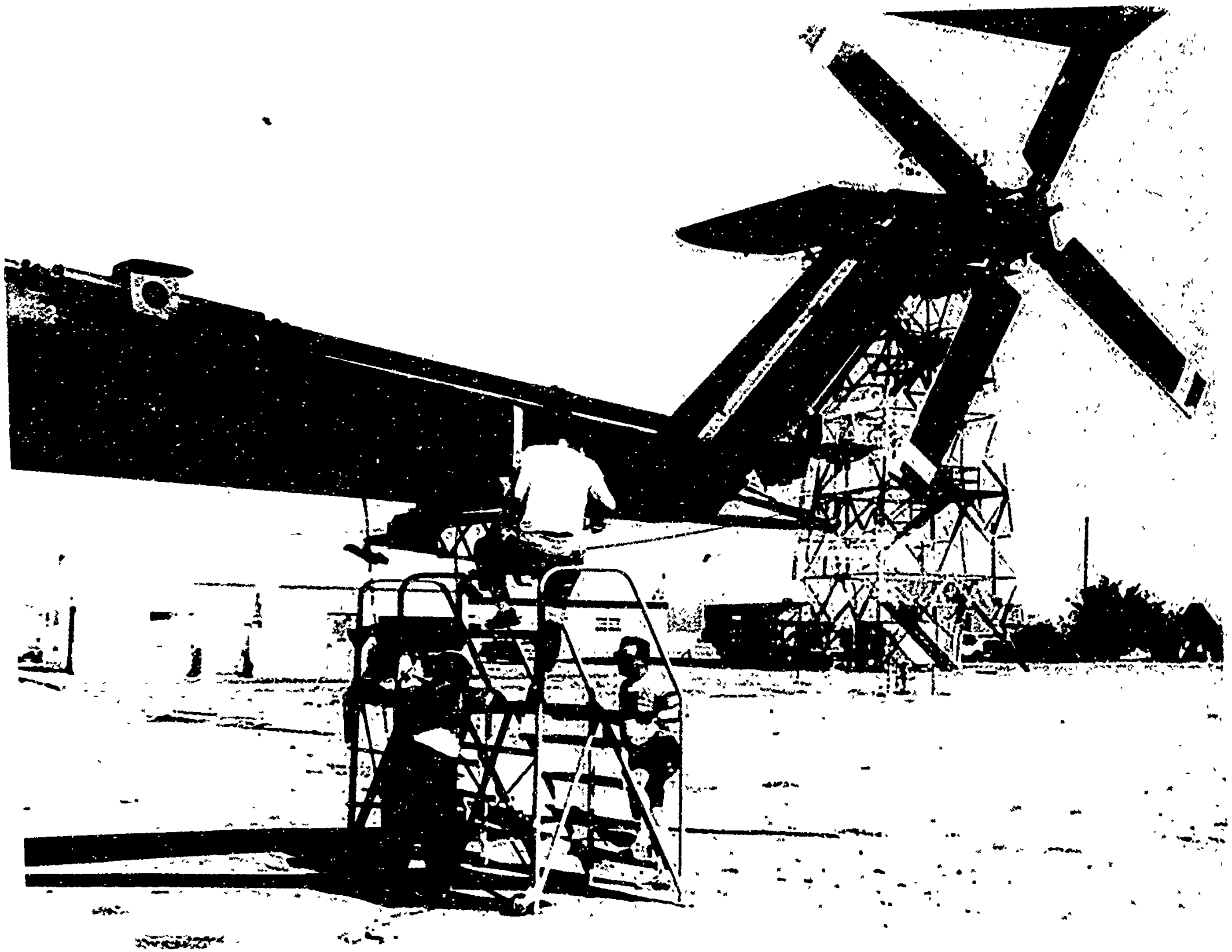

Figure 31. Installation of ESD Exciters and Multipliers at Fuselage Station 704.

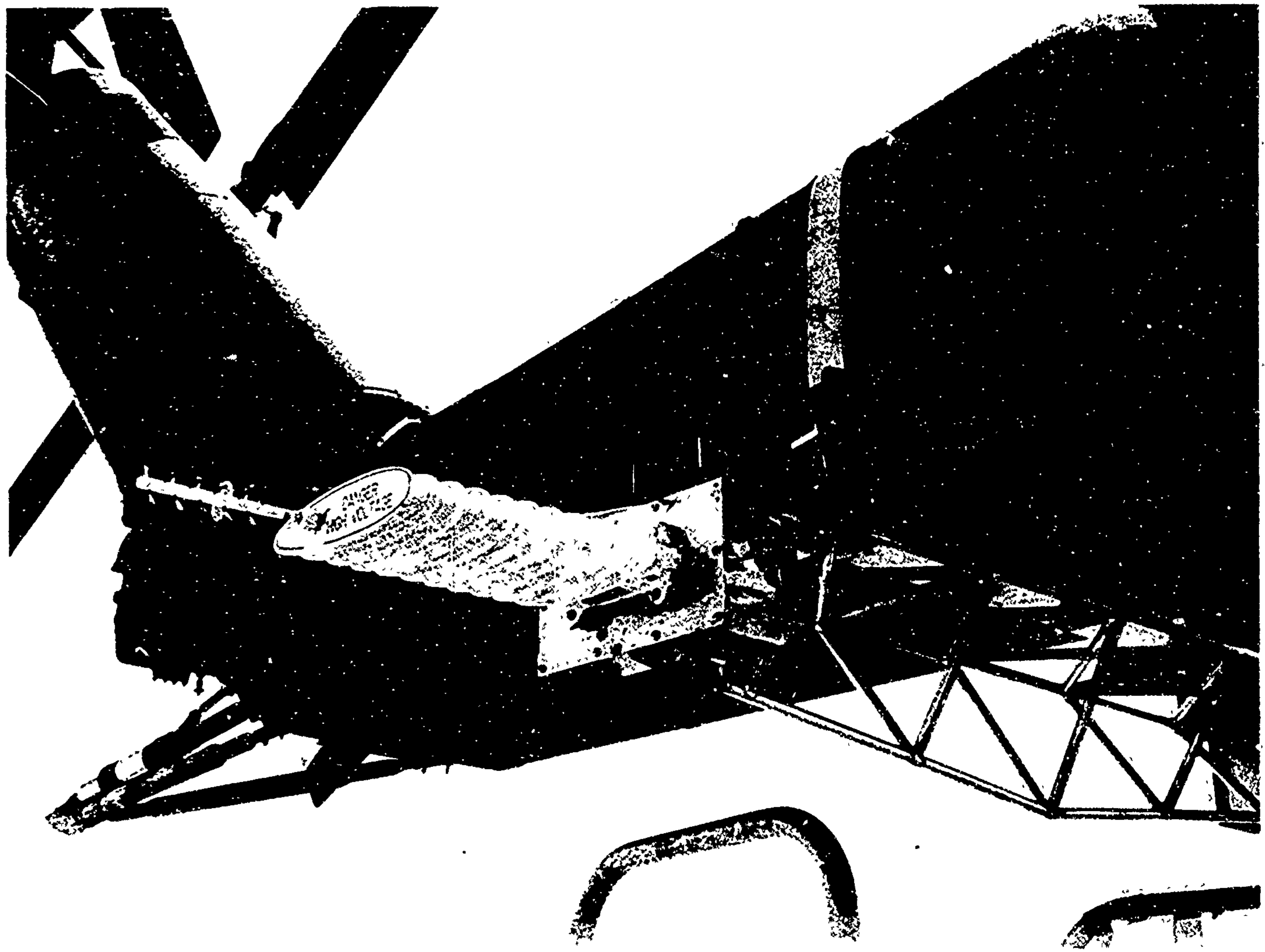

Figure 32. Detail of Exciter/Multiplier Test Installation at Fuselage Station 704.

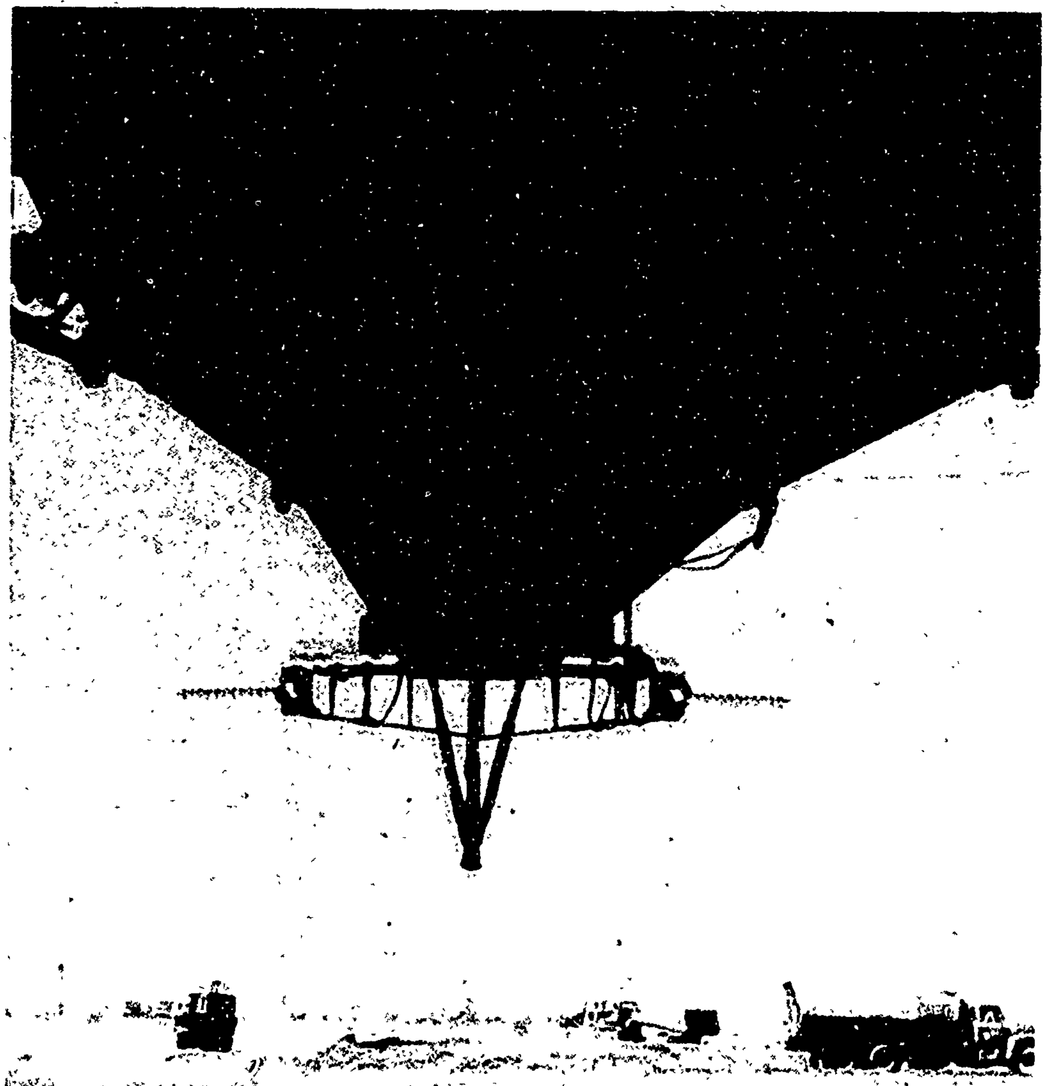

Figure 33. Completed Exciter/Multiplier Installation on Tail Cone of CH-54A Test Aircraft at Fuselage Station 704.

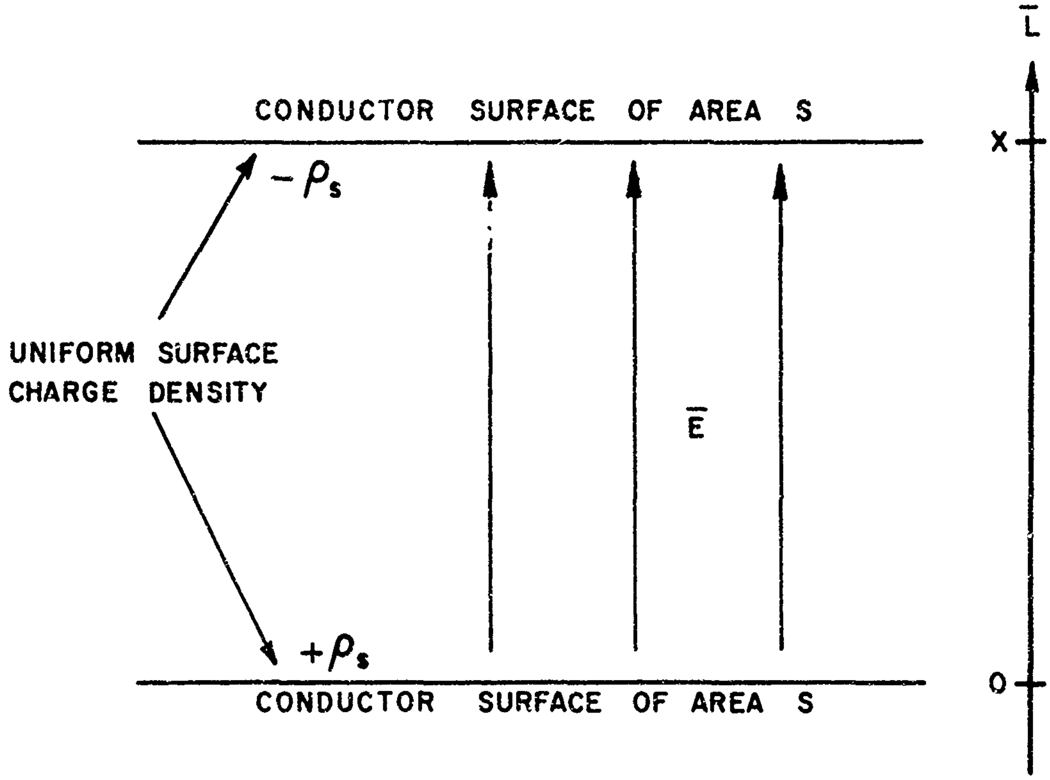

Figure 34. Idealized Parallel-Plate Capacitor.

S, which are separated by a distance x. The intra-plate medium has a permitivity, or inductive capacity, of ϵ. The potential difference V between the lower and upper plates is

$$V = -\int_{upper}^{lower} E \cdot dL \quad \text{(the scalar product of two vectors)} \tag{24}$$

$$V = -\int_{x}^{0} \frac{\rho_s}{\epsilon}\, dL \tag{25}$$

$$V = \frac{\rho_s x}{\epsilon} \tag{26}$$

where ρ_s is the surface charge density on the plates.

The electric field and charge distribution are assumed to be almost uniform at all points not adjacent to the edges, and this latter region is assumed to contribute only a small percentage of the total capacitance. Therefore, the total charge Q on the capacitor is

$$Q = \rho_s S \tag{27}$$

The capacitance is

$$C = Q/V \tag{28}$$

$$C = \frac{\rho_s S}{\frac{\rho_s x}{\epsilon}} \tag{29}$$

$$C = \epsilon S/x \tag{30}$$

Relating this back to the situation of a hovering helicopter, equation (30) states that the capacitance will decrease as the hovering altitude x increases.

The capacitance test situation is illustrated in Figure 35. The aircraft/ground capacitance C_A can be determined in much the same manner as can that of any other capacitor in the picofarad range; that is, through the use of an impedance bridge. Because of the relationship shown in equation (30), it is desirable to plot the capacitance as a function of altitude. The bridge is located on the aircraft, and a ground connection is made through the drop line. However, as indicated in Figure 35, the bridge measures the parallel sum of the aircraft/ground capacitance and a spurious capacitance C_S between the aircraft and the drop line. A second measurement, made with the drop line disconnected from ground, isolates C_S. Subtracting C_S from the total capacitance yields the desired C_A reading.

NATURAL CHARGING CURRENT TEST

The test setup used to measure natural charging current is illustrated in Figure 36. The charge is shown entering the aircraft at the rotor blades, at a rate I_N. This is the principal mode of triboelectric charging. The aircraft is shorted to ground through a microammeter, by means of the drop line. The microammeter reads the drop line current, I_{DL}. Since the aircraft-to-ground voltage V_A is zero, I_{DL} is equal to I_N, the natural charging current. With reference to the polarity convention discussed above, a positive value of I_{DL} corresponds to a negative value of I_N.

DISCHARGING CAPABILITY TEST

The purpose of the discharging capability test is to obtain data regarding the efficiency of the high-voltage multipliers as dischargers.

For this test, the system is operated in an open-loop mode. In fact, a test plate is placed over the sensor head so that it does not read the aircraft charge. Instead, an input signal, which is generated at the ESD test set, is applied to the test plate. This input signal is varied to control the output; namely, the probe current from the multiplier. As was the case during the natural charging current measurements, the aircraft is shorted to ground through a microammeter (see

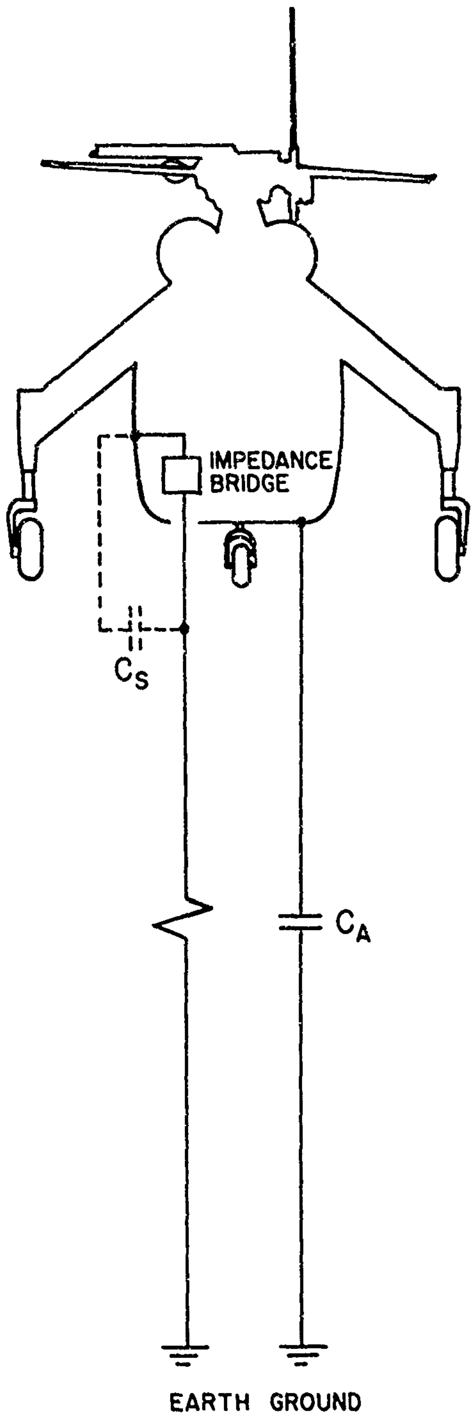

Figure 35. CH-54A Configuration for Aircraft/Ground Capacitance Measurements.

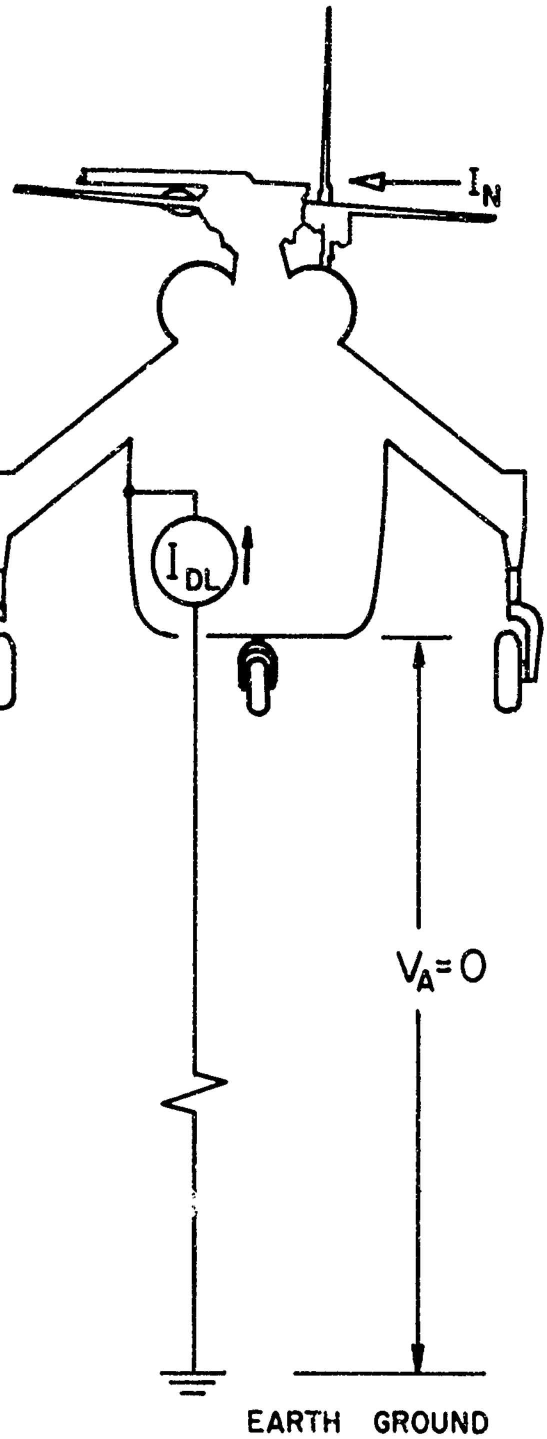

Figure 36. CH-54A Test Configuration for Natural Charging Measurements.

Figure 37). The drop line current is equal to the sum of the ESD discharge current I_{DE} and the natural charging current I_N. With the polarities indicated in Figure 37,

$$I_{DL} = I_{DE} - I_N \tag{31}$$

It is generally advisable to conduct this test when the natural charging current is small because higher currents tend to vary considerably over short periods of time due to changes in ambient wind, aircraft attitude, etc. The charging rate experienced by a helicopter hovering over a paved ramp at an altitude of about 25 feet is sufficiently small that it may be neglected in considering this test setup. Therefore,

$$I_{DL} = I_{DE} \tag{32}$$

The ESD output is the probe current I_P. Ideally, and optimally, $I_P = I_{DE}$. But, when there is recirculation,

$$I_P = I_{DE} + I_R \tag{33}$$

Therefore,

$$I_R = I_P - I_{DE} \tag{34}$$

By substituting,

$$I_R = I_P - I_{DL} \tag{35}$$

Both I_P and I_{DL} are measured during the test. The technique is to adjust the input signal for discrete values of I_P and read I_{DL}. In terms of the polarity convention, a positive value of I_{DL} corresponds to a positive value of I_{DE}.

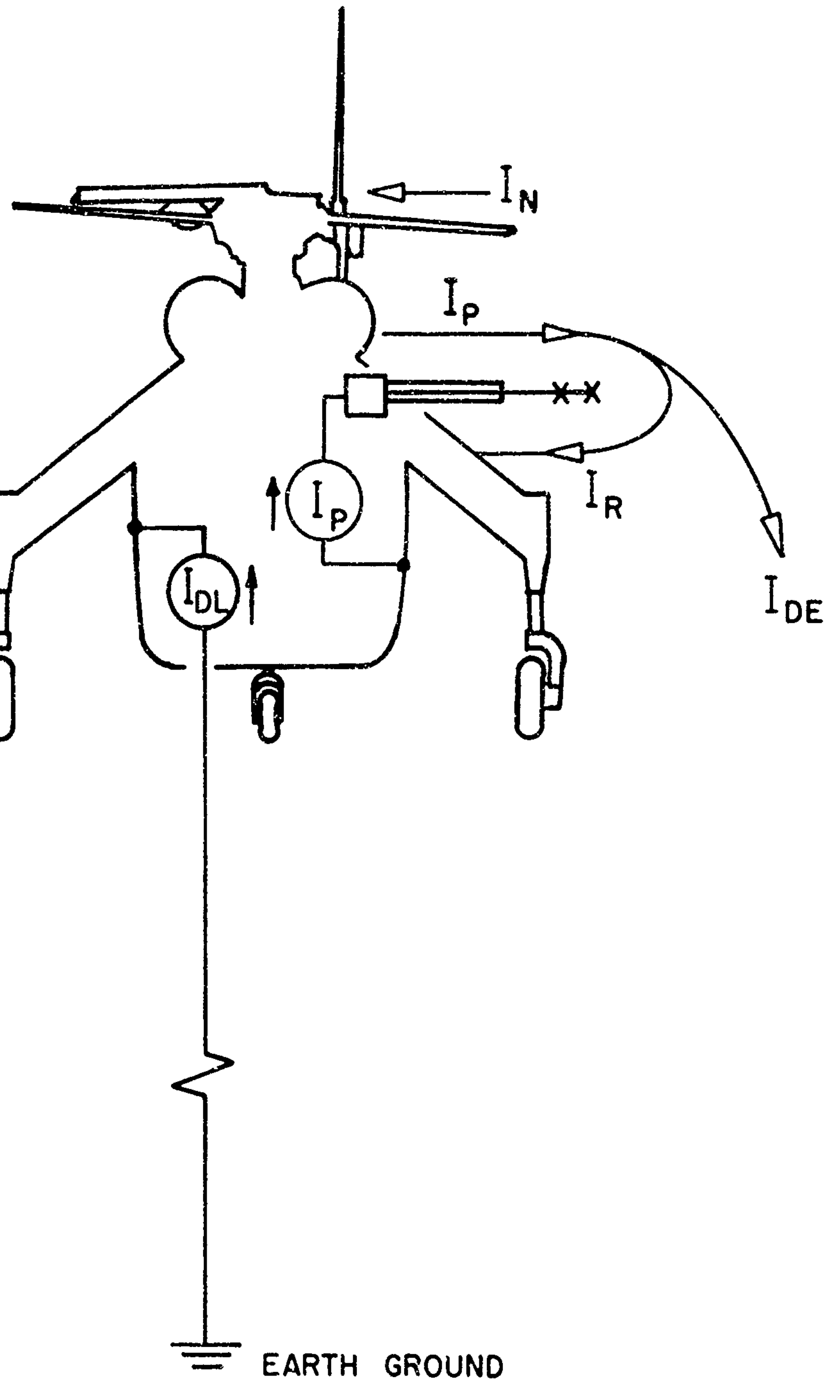

Figure 37. CH-54A Test Configuration for Measurement of ESD System Discharging Capability.

A sample of the data collection form used at Yuma to facilitate recording of the necessary readings is shown in Figure 38.

TEST OF ESD OPERATION UNDER NATURAL CHARGING CONDITIONS

Figure 39 pictures the testing of the ESD in a natural charging situation. The charge is shown entering the helicopter at the rotor blades at a rate I_N. The natural charging current is determined in the manner described above and illustrated in Figure 36. The ESD operates to discharge the aircraft at the same rate, so that the probe current I_P is equal in magnitude to I_N (neglecting any possible recirculation). A small residual charge Q_A remains on the aircraft to provide a driving force for the discharger output.

The magnitude of Q_A can be inferred from the aircraft-to-ground potential V_A which is measured on the EVM. The ESD test set, located in the aircraft cockpit, provides a readout of I_P.

DISCHARGING CAPABILITY TEST DATA SHEET

I_{Probe} (Microamperes)	$I_{Drop\ Line}$ (Microamperes)	$I_{Recirculation}$ (Microamperes)	Discharging Efficiency (Percent)

ESD Model

E/M Location ____________

Date ____________

Remarks ____________

Figure 38. Data Collection Form Used During Discharging Capability Tests.

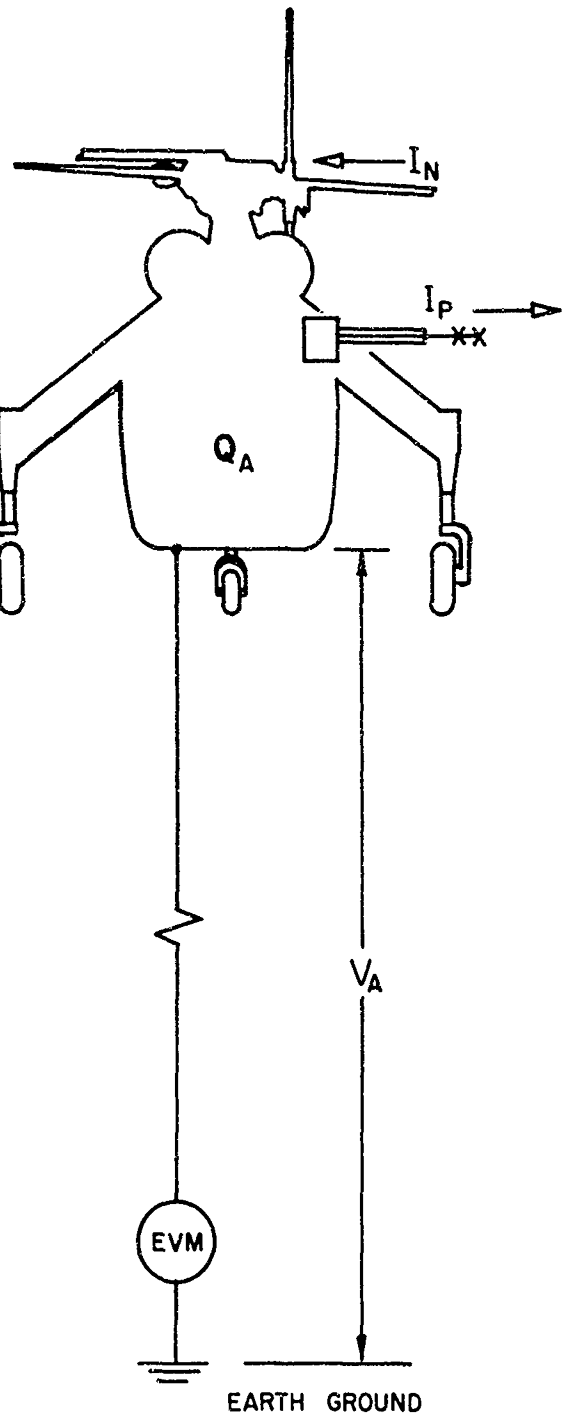

Figure 39. CH-54A Test Configuration for Measurement of Aircraft Potential With ESD System in Operation.

APPENDIX II
GRAPHS OF DATA

Figures 40 through 44 are plots of the discharging capability test data for the ESD systems under various conditions of exciter/multiplier location, as indicated.

With one exception, the natural charging current has been ignored in plotting these curves. The reason for this is the fact that the natural charging current of a "light" CH-54A hovering over a paved ramp is minimal; that is, on the same order as the readability of the instrumentation used.

However, the data used in plotting Figure 43 were recorded while the aircraft was carrying the 6000-pound load and, as a consequence, experiencing a significant natural charging current. This curve is plotted on the basis of a 10-microampere positive natural charging current.

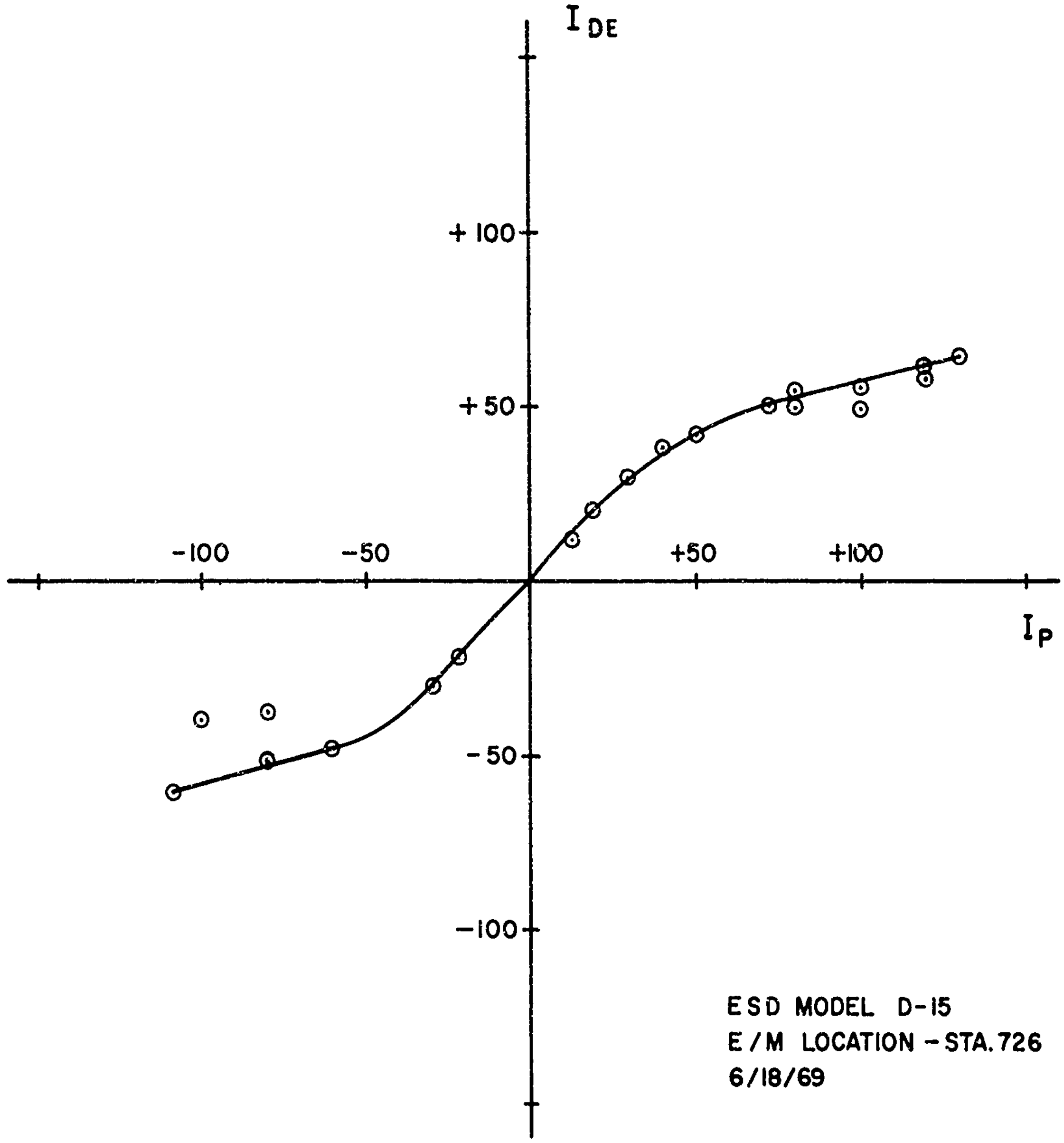

Figure 40. Discharging Capability Test Data.

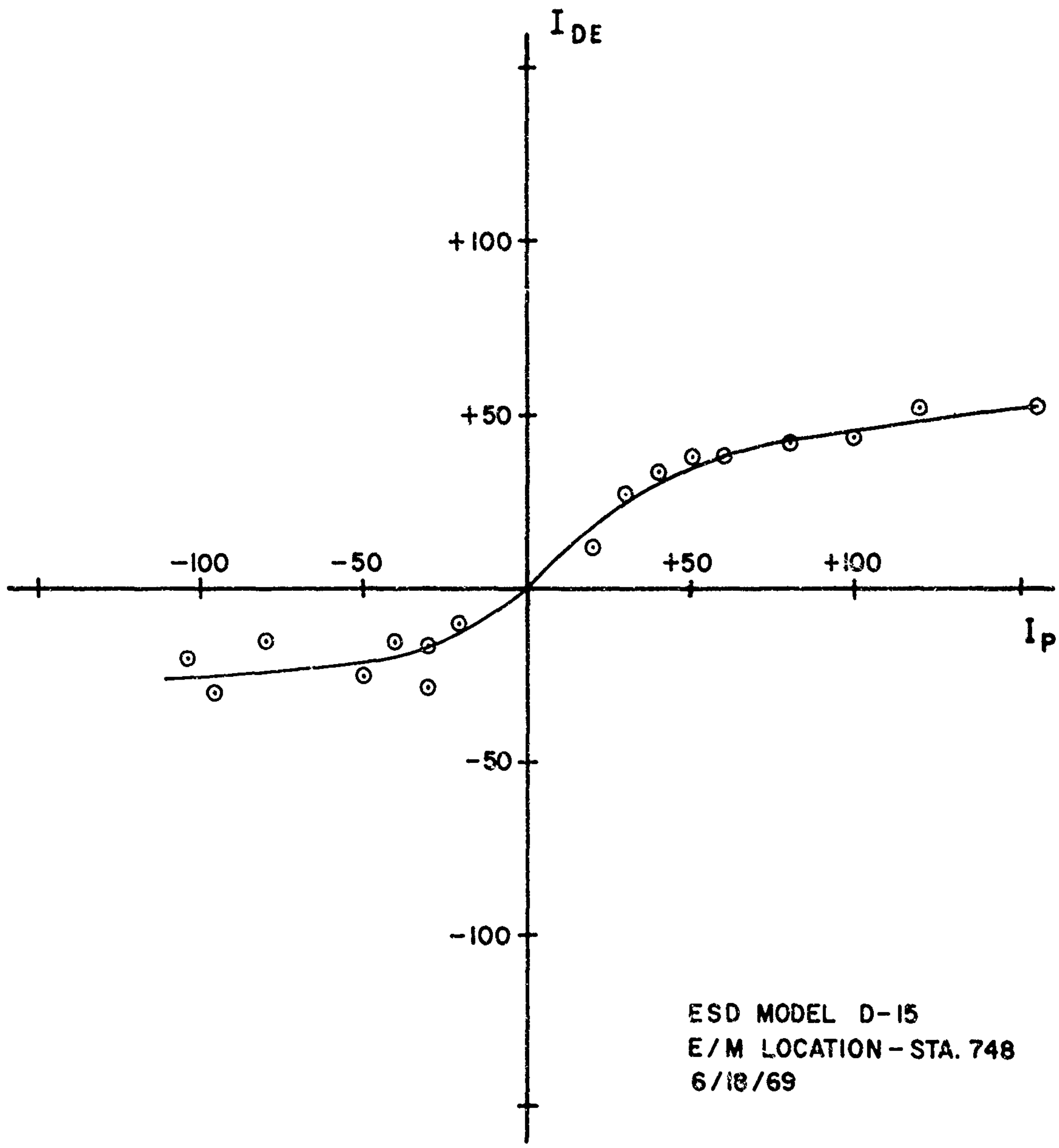

Figure 41. Discharging Capability Test Data.

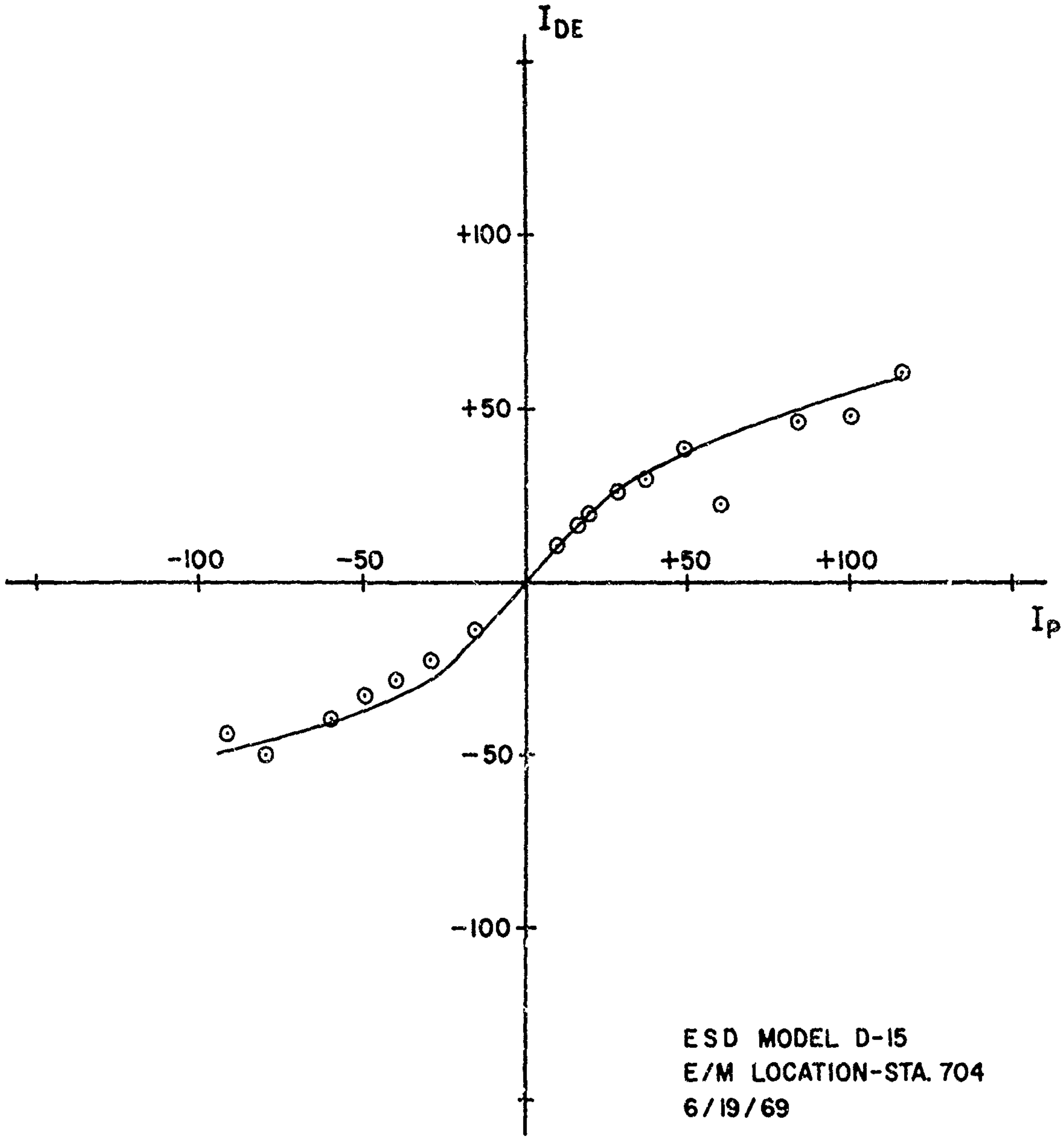

Figure 42. Discharging Capability Test Data.

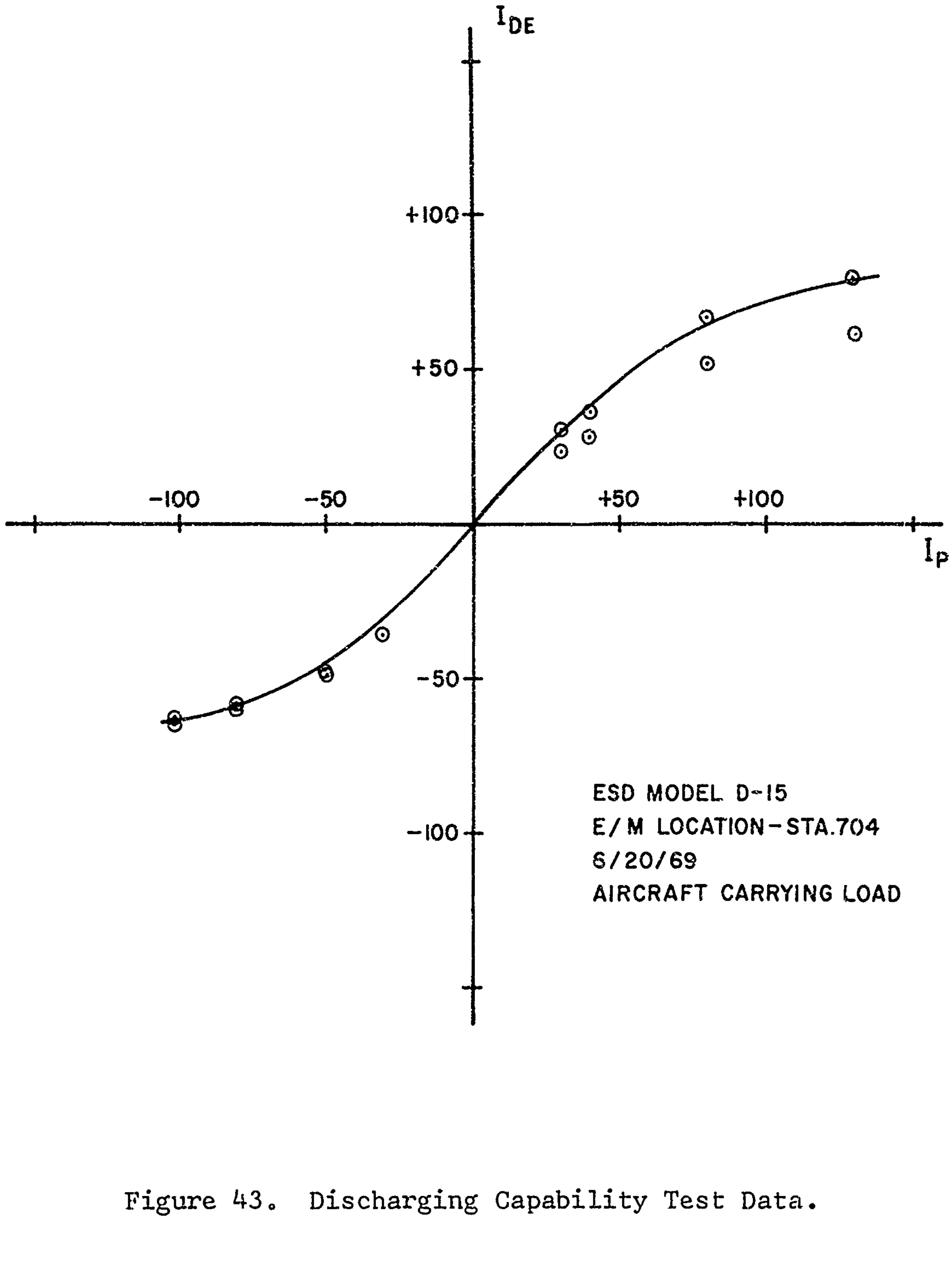

Figure 43. Discharging Capability Test Data.

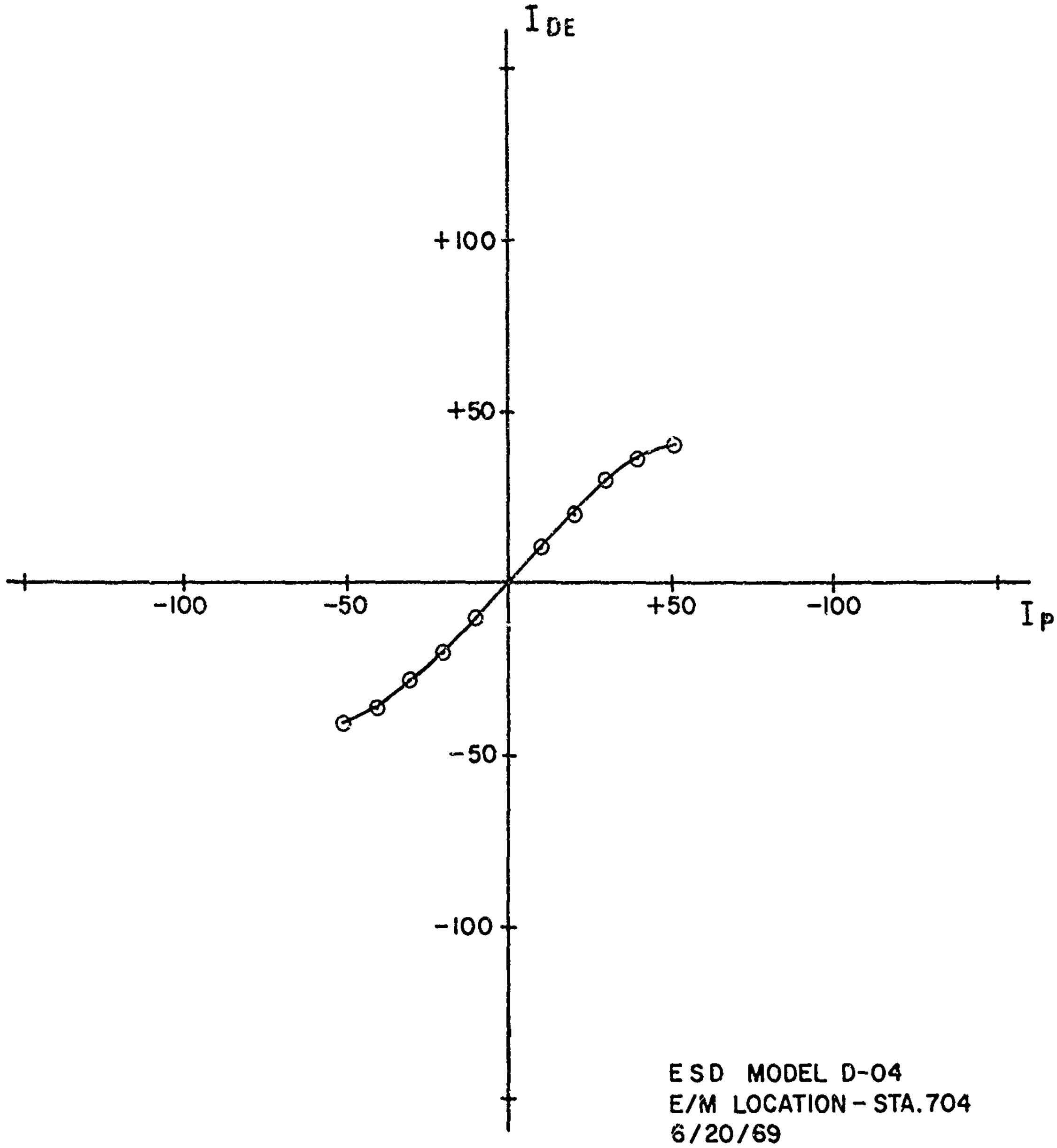

Figure 44. Discharging Capability Test Data.

APPENDIX III
PROBLEMS ASSOCIATED WITH
ELECTROSTATICALLY CHARGED HELICOPTERS

GENERAL

Helicopters in flight can accumulate a sizable electrostatic charge. Experience has indicated three main problem areas associated with electrostatically charged helicopters. Field personnel performing underslung cargo operations with electrostatically charged aircraft have been subjected to severe electrical shocks while attaching cargo slings to helicopter cargo hooks. The danger of exploding fuel and munitions cargo due to spark contact during external hookup operations has also been a major source of concern. In addition, electrostatic charge on aircraft can result in radio frequency interference (RFI) on communications and navigation systems.

TRIBOELECTRIC EFFECTS

The primary cause of helicopter charging is triboelectric transfer between the aircraft and the air through which it moves. Triboelectric charging is generated by bringing two dissimilar substances into contact; in this case, the substances are particles of air, dust, sand, snow, or rain in moving contact with the helicopter rotor blades. The rate of charge influx (natural charging current) is determined by the number and types of charges impinging upon the aircraft, which, in turn, are related to ambient conditions, such as weather and terrain, and the size and geometry of the helicopter. The polarity of charge that accumulates on the aircraft is determined by the relative dielectric constants of the aircraft and the particles impinging upon it.

AIRCRAFT CAPACITANCE

A finite capacitance exists between a hovering helicopter and ground. The value of this aircraft-to-ground capacitance is dependent upon the aircraft shape, size, and proximity to ground for altitudes of less than about 25 feet. This capacitance is, by definition, a measure of the aircraft's ability to store electric charge.

AIRCRAFT POTENTIAL

In the process of storing charge, the helicopter develops a voltage with respect to ground. This voltage V is related to the aircraft-to-ground capacitance C and the total charge on the aircraft Q by the classical formula $V = Q/C$. Because the aircraft has a potential with respect to its surroundings, corona discharge will take place from the helicopter when the breakdown field of the air is exceeded. A steady-state condition will be reached, wherein the corona discharge current is equal to the natural charging current. The steady-state residual aircraft voltage will be that required to support corona discharge at a rate equal to that of charge influx. This is typically on the order of 30,000 to 200,000 volts, depending upon the particular helicopter type and the ambient natural charging rates.

STORED ENERGY

When a helicopter is charged, energy is stored thereon. The formula $E = \frac{1}{2}CV^2$ expresses the relationship between the aircraft capacitance and voltage and the total energy available for discharge when the aircraft is grounded.

PROBLEM ANALYSIS

The causes of the problems associated with charged aircraft are apparent from the preceding paragraphs.

Shock Hazard

The release of energy stored on the aircraft is the cause of the shock or spark hazards.

Any discussion of personnel shock hazard must make a distinction between electrostatic shock hazard and the shock hazard associated with an AC or DC power source. In the case of shock from electrostatic discharge, the determinant for the severity of the shock is the amount of energy discharged (see Figure 45); whereas, when the shock is associated with a power source, the measure of the shock's intensity lies in the amount of current forced through the body (see Figure 45).

Electrostatic discharge, of itself, is dangerous at a level of about 1 joule. However, it is somewhat below this level that the energy transfer to the body causes muscle spasms and the resultant sensation of being physically thrown. The energy transfer becomes uncomfortable at a much lower level, 10 millijoules on the average; the resultant shock is considered to be severe at a level of 100 millijoules. The severity of personal injury at any given energy level is a function of individual health. General statements relating danger to average energy levels are widely accepted.[3] The

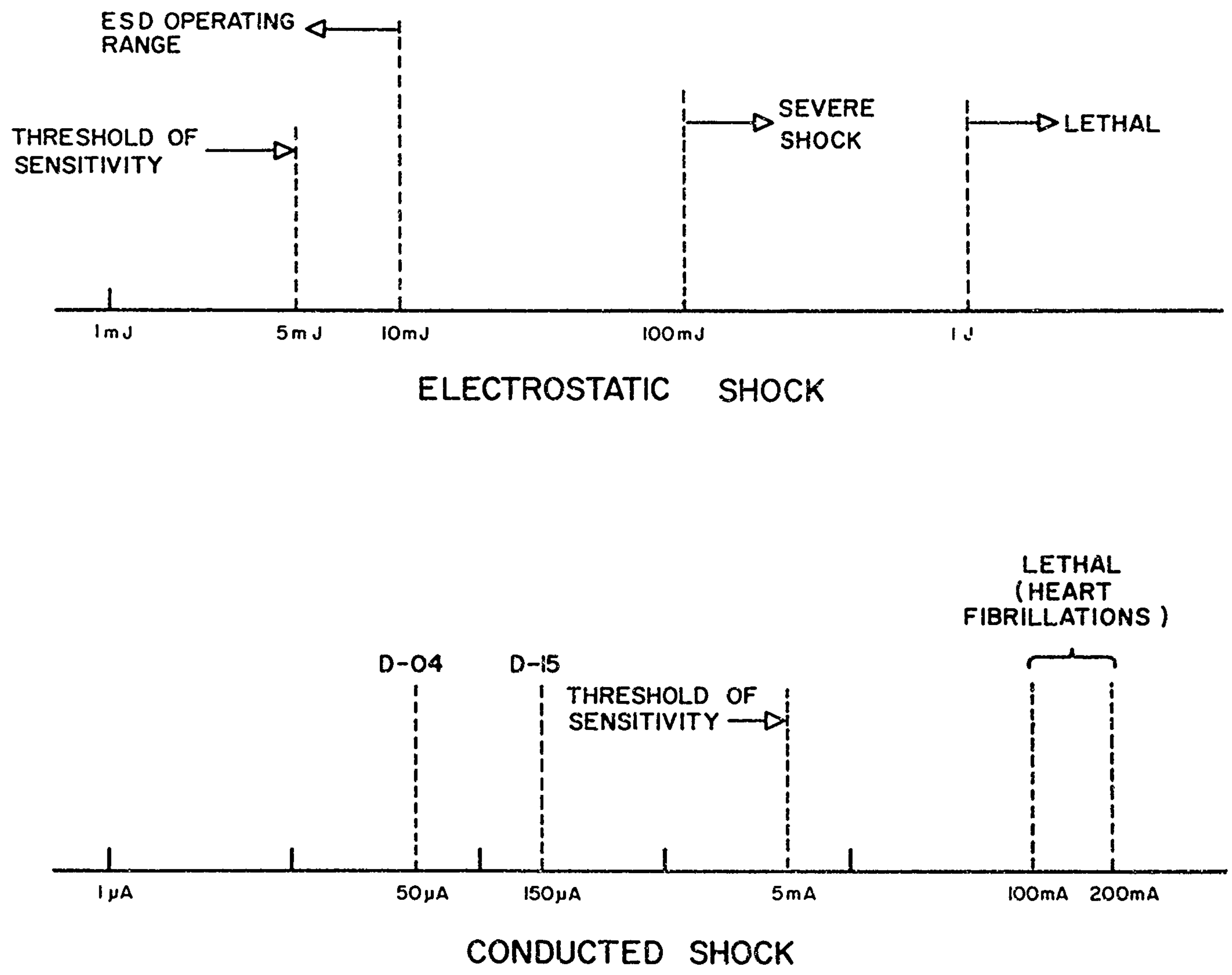

Figure 45. Personnel Shock Hazard Levels.

threshold of sensitivity, that is, the energy level at which electrostatic discharge is just noticeable, averages about 5 millijoules. (These average figures are based upon results of tests performed by Dynasciences in August 1968.) This threshold varies depending upon circumstances; it is somewhat lower for a person standing in water or on a grounded metal plate and higher for an individual standing on concrete or wood.

Energy levels of 0.64 to 0.88 millijoule will ignite aviation fuels under conditions of most ideal temperature and air mixture for combustion.[6] However, for the conditions under which these substances are encountered while sling loading, permissible energy levels are considerably higher.

The severity of a shock from a power source, as indicated above, is dependent upon the amount of current forced through the body.

The threshold of sensitivity for shock of this nature is 5 milliamperes. Any amount of current over 10 milliamperes is capable of producing a painful to severe shock; at values as low as 20 milliamperes, breathing becomes labored, finally ceasing completely at about 75 milliamperes. Currents between 100 and 200 milliamperes are lethal. As the current approaches 100 milliamperes, it causes ventricular fibrillation of the heart; that is, an uncoordinated twitching of the ventricle walls. Above 200 milliamperes, the muscular contractions are so severe that the heart is forcibly clamped during the shock, and the victim's chances for survival are good. Currents above 5 amperes are fatal because of the heating due to the power, I^2R, dissipated in the body.

Interference

High electrostatic charge can also generate RFI. Uncontrolled electrostatic discharge, or corona, due to a high aircraft potential will occur from any relatively sharp edge or point on the aircraft. The voltage required to initiate corona is proportional to the radius of the edge or point at which corona occurs. Antennas are frequently among the first parts of the aircraft to go into corona. This is one of the causes of the familiar "static" on associated radio equipment.

APPENDIX IV
BASIC CONSIDERATIONS FOR THE STATISTICAL ANALYSIS OF TEST DATA

PARAMETERS OF A SAMPLE

Given a set of data (a sample), any quantity which can be determined on the basis of the sample values $x_1, x_2, \ldots x_n$ is called a statistic. The sample mean is, therefore, a statistic. This is a value assumed by the random variable $\bar{x}$, where

$$\bar{x} = \sum_{i=1}^{n} \frac{x_i}{n} \qquad (36)$$

Another important statistic is the sample variance, which is the most widely used measure of the variability of a sample. This is a value assumed by s^2, where

$$s^2 = \frac{\sum_{i=1}^{n} (x_i - \bar{x})^2}{n-1} \qquad (37)$$

Its positive square root s is called the sample standard deviation.

PARAMETERS OF A NORMAL POPULATION

The distributions of $\bar{x}$ and s^2 are referred to as sampling distributions. If the data points are random samples from a normal population--an infinite population of the form $N(x_i; \mu, \sigma^2)$--the independent random variables, whose values constitute the random sample, have identical normal distributions with the population mean μ and the population variance σ^2. If $\bar{x}$ is the mean of a random sample of size n from the normal population $N(x; \mu, \sigma^2)$, then the sampling distribution of $\bar{x}$ is the normal distribution $N(\bar{x}; \mu, \sigma^2/n)$. In addition, a theorum of applied statistics states that, if s^2 is the variance of a random sample of size n from the normal population $N(x; \mu, \sigma^2)$, then $(n-1)s^2/\sigma^2$ has a chi-square distribution with n-1 degrees of freedom.

USE OF SAMPLE PARAMETERS IN DECISIONS ABOUT A POPULATION

Through the process of statistical inference, it is possible to arrive at conclusions or decisions concerning the parameters of a population on the basis of information contained in samples. For instance, μ, the mean of a population, can be estimated on the basis of the sample mean $\bar{x}$. Likewise, the population variance σ^2 can be estimated from $\bar{x}$ and the sample variance s^2.

The most useful estimates are those of intervals and, more specifically, confidence intervals for the parameters of the population. Basically, the idea of a confidence interval is the assertion, with a certain stated probability, that the value of the random variable falls within a stated interval.

For a random sample of size n from a normal population, $(\bar{x}-\mu)\sqrt{n}/s$ has a t distribution with n-1 degrees of freedom. Therefore, it can be asserted with a probability of $1-\alpha$ that this random variable will assume a value between $-t_{\alpha/2,\ n-1}$ and $+t_{\alpha/2,\ n-1}$. Making use of this characteristic, it is possible to construct a $1-\alpha$ confidence interval for μ when σ is unknown. For a given sample, a degree of confidence of $1-\alpha$ can be assigned to

$$-t_{\alpha/2,\ n-1} \leq (\bar{x}-\mu)\sqrt{n}/s \leq t_{\alpha/2,\ n-1} \tag{38}$$

or

$$\bar{x} - t_{\alpha/2,\ n-1}\frac{s}{\sqrt{n}} \leq \mu \leq \bar{x} + t_{\alpha/2,\ n-1}\frac{s}{\sqrt{n}} \tag{39}$$

Given a random sample of size n from a normal population, a $1-\alpha$ confidence interval for σ^2 may be obtained by using $(n-1)s^2/\sigma^2$, which has a chi-square distribution with n-1 degrees of freedom. It can thus be asserted with a probability of $1-\alpha$ that this random variable assumes a value between $\chi^2_{1-\alpha/2,\ n-1}$ and $\chi^2_{\alpha/2,\ n-1}$ or with a degree of confidence of $1-\alpha$ that for a given sample

$$\chi^2_{1-\alpha/2,\ n-1} \leq \frac{(n-1)s^2}{\sigma^2} \leq \chi^2_{\alpha/2,\ n-1} \tag{40}$$

or

$$\frac{(n-1)s^2}{\chi^2_{\alpha/2,\ n-1}} \leq \sigma^2 \leq \frac{(n-1)s^2}{\chi^2_{1-\alpha/2,\ n-1}} \qquad (41)$$

This $1-\alpha$ confidence interval for σ^2 may be converted into a $1-\alpha$ confidence interval for σ by taking square roots.[7]

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13. ABSTRACT

A test program was conducted at Yuma Proving Ground, Arizona, to obtain in-flight measurements of electrostatic charging rates experienced by a CH-54A helicopter operating in a dusty environment and to evaluate active electrostatic discharger systems as solution techniques. This report presents data obtained during this program and reviews these data along with those from previous work on the same aircraft type. This report concludes that an extremely high and possibly lethal charge level is present on the CH-54A in the operational hookup situation, that an active discharger system capable of discharging 200 microamperes is required to dissipate this charge, and that, while presently available equipment does not meet this requirement, repackaging of present hardware offers a probable solution.

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	ROLE	WT	ROLE	WT	ROLE	WT
Electrostatic Static Charges, Static Charge Electrostatic Dischargers Electrical Charges P-Stats ESD Triboelectric Charging Atmospheric Electricity CH-54 Null Field Dischargers (NFD) Wicks (Electrostatic) Probes (Discharge) Active Dischargers Active Electrostatic Discharge Systems Potentials, Electostatic Precipitation Static Lightning Discharge Shock External Cargo Hookup Operations Energy, Potential Capacitance Natural Charging Artificial Charging Earth Field Electrostatic Field Dust Engine Charging						

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